1969-10

The extension of an adaptive aircraft control concept to helicopters.

Hood, John McCoy Jr.

Monterey, California. U.S. Naval Postgraduate School

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by

John McCoy Hood, Jr.

United States
Naval Postgraduate School

THESIS

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October 1969

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John McCoy Hood, Jr.
Lieutenant, United States Navy
B.S., United States Naval Academy, 1963

Submitted in partial fulfillment of the
requirements for the degree of

MASTER OF SCIENCE IN AERONAUTICAL ENGINEERING

from the

NAVAL POSTGRADUATE SCHOOL
October 1969

ABSTRACT

A basic adaptive control scheme for fixed-wing aircraft was modified for use in controlling the longitudinal motion of helicopters. The modification required the addition of two additional feedback variables. Control was applied only to the cyclic pitch input and not to the collective input. It was assumed that a coefficient, the cyclic-pitch control effectiveness, would not change sign throughout the flight envelope.

Analog computer simulation showed that the modified system was capable of stabilizing the model used. The handling qualities of the system were not completely satisfactory and further work is necessary.

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Capital Letters

B = A time dependent coefficient

C = Pilot input to cyclic; positive forward

C^* = Sum of normal and tangential accelerations felt by the pilot plus constant multiples of pitch attitude and pitch rate

K = Fixed positive gain

M = Moment about the Y axis; positive up

M_u, etc. = Partial derivative of M with respect to u, etc.

U_o = Initial forward velocity along the X axis

V = Non-negative error parameter

W_o = Initial velocity along the Z axis; positive down

X = Force along the X axis; positive forward

X_u, etc. = Partial derivative of X with respect to u, etc.

Z = Force along the Z axis; positive down

Z_u, etc. = Partial derivative of Z with respect to u, etc.

Lower Case Letters

a = Tangential acceleration along the X axis; positive forward

f = Cyclic feedback

g = Acceleration of gravity

n = Normal acceleration along Z axis; positive up

q = Perturbation pitch rate (also $\dot{\theta}$)

s = First derivative with respect to time

s^2 = Second derivative with respect to time

u = Perturbation velocity along the X axis; positive forward

TABLE OF SYMBOLS
(continued)

$\dot{u}$, etc.	=	First derivative with respect to time, etc.
w	=	Perturbation velocity along the Z axis; positive down

Greek Letters

β	=	Perturbation blade-flapping angle
β	=	Fixed parameter
$\bar{\beta}$	=	Variable parameter based on stability derivatives
β_u, etc.	=	Partial derivative of β with respect to u, etc.
Γ	=	Variable gain
Γ^*	=	Ideal steady state gain
δ_θ	=	Cyclic pitch input to rotor; positive forward
δ_L	=	Collective pitch input to rotor; positive up
ϵ	=	Error
θ	=	Peturbation pitch attitude
$\dot{\theta}$, etc.	=	First derivative of θ with respect to time, etc.
$\ddot{\theta}$, etc.	=	Second derivative of θ with respect to time, etc.

ACKNOWLEDGMENT

It is fitting to hereby acknowledge the assistance of the people who contributed to the successful completion of this study. The author wishes to express his appreciation to Mr. Robert Limes for his assistance during the analog computer phase of the evaluation; and to Mr. Warren Woodford for his assistance with the graphics work.

The author is particularly indebted to Associate Professor Donald Layton for his assistance during the write-up phase and to Professor Edward Rang for his guidance and assistance throughout the investigation; and to his wife, Ginger, for her assistance in preparing this manuscript and for her patience during the entire course of studies.

I. INTRODUCTION

The technology of larger, faster and more complex aircraft has been increasing over the past few years at a tremendous rate. Although the development of the fast or large fixed-wing aircraft such as the F-111 and C-5 have captured the major headlines, the advance in helicopter technology has been equally remarkable. The new breeds of aircraft have required the design of more powerful powerplants to fly the machines higher, faster and for longer periods of time. New structural materials, designed to withstand the higher stresses and temperatures imposed, have been required. As aircraft mission and complexity have increased, new electronic systems have been designed to aid the pilot. The design of more reliable and complex automatic flight control systems (AFCS) has been required.

Helicopters are normally very unstable, especially at low speeds where they accomplish the major portions of their missions. The increased complexity of these missions makes a highly reliable AFCS an absolute necessity. Older helicopters were severely limited in both payload and endurance and were successfully operated by pilot skill alone. The advent of large helicopters, capable of all-weather night operations with long endurances, made it nearly impossible for the pilot to fly the aircraft without the aid of a system to augment stability. Although the pilot must be able to operate a helicopter without such compensation, flight under these conditions must be considered to be close to emergency operation. For example, the NATOPS Flight Manual for Navy Model SH-34J helicopters, an aircraft of relatively low complexity, requires that the automatic stabilization

equipment be operating prior to any night or instrument flight. It should be noted that the H-34 has been in service for a number of years and is now being replaced by far more complex helicopters.

Most automatic flight control systems now in use require the measurement of air data, such as airspeed, altitude, angle of attack, etc., in order to accomplish an elaborate gain scheduling over the entire range of flight conditions. In contrast to this, an adaptive controller uses direct measurement of the aircraft responses, such as pitch attitude and accelerations, to automatically compute the gains required at the present flight condition. Operation of such a system might be compared to the operation of the human body in that the body is able to adapt itself to new conditions, such as changes in temperature or altitude, so that it maintains certain desired parameters within desired limits.

Although the F-111 is the only production aircraft presently using an adaptive controller, much effort has gone into developing the adaptive system to a point where it will be more economical and reliable than the present systems using air data measurement. Several modifications and refinements to the basic developments of Shipley, et. al., (Ref. 1) are reported in Refs. 2-4. These offer new hope that development of a superior adaptive controller will be forthcoming.

Although these previous investigations of adaptive control schemes were limited to fixed-wing aircraft, it appeared that such a controller would be equally well suited for helicopters.

The instability and difficulty in measuring air data at low speeds offer an opportunity to test the flexibility of the adaptive system theory previously developed. Because of the instability, and addition

of collective control, the resulting controller would have to be more complex than a similar system used in fixed-wing aircraft. It was hoped that the system would not only stabilize the aircraft at all flight conditions, but would also yield desirable handling qualities. The handling qualities criteria as presented in Ref. 5, section 4.3, were given particular attention in developing the system.

The approach presented in Ref. 2 was used as a basis for the system. Reference to the nonvarying-C^* criterion and addition of servo and actuator problems were not considered in the initial analysis. It was hoped however that the C^* criterion would be adapted for helicopter use. Basic considerations of the adaptation are presented in Chapter II.

Derivation of the required equations and the application of the equations to the analog computer is presented in Chapter II. Results of tests conducted on the computer are contained in Chapter III. Conclusions drawn from the tests conducted and recommendations for further investigations are given in Chapter IV.

II. ADAPTATION OF THE SYSTEM FOR USE IN HELICOPTERS

A. DERIVATION OF EQUATIONS:

The adaptive control scheme based on the short-period perturbation equations used in Ref. 2 was not acceptable for helicopter use because of the importance of the phugoid mode. It was necessary to use the full set of perturbation equations, which complicated the problem by adding additional variables. Using the assumptions of constant rotor speed and of no coupling between the longitudinal, lateral, rolling and yawing moments as outlined in Ref. 6, the longitudinal perturbation equations of motion are

Drag:
$$(S-X_u)u - X_w w + (W_0 S + g)\theta - (X_{\dot\beta} S + X_\beta)\beta = X_\delta \delta_\theta + X_L \delta_L \qquad (1)$$

Lift:
$$Z_u u - (S-Z_w)w + U_0 S\theta + (Z_{\dot\beta} S + Z_\beta)\beta = -Z_\delta \delta_\theta - Z_L \delta_L \qquad (2)$$

Moment:
$$M_u u + (M_{\dot w} S + M_w)w - (S^2 - M_q S)\theta + (M_{\dot\beta} S + M_\beta)\beta = -M_\delta \delta_\theta - M_L \delta_L \qquad (3)$$

Blade flapping:
$$\beta_u u + \beta_w w + \beta_q S\theta + (\beta_{\dot\beta} S + \beta_\beta)\beta = -\beta_\delta \delta_\theta - \beta_L \delta_L . \qquad (4)$$

In order to simplify the problem for initial analysis the following assumptions were made.

1. Initial level flight ($W_0=0$)

U_o - Initial Forward Velocity

u, w - Perturbation Velocities

θ - Pitch Angle

$\dot{\theta}$ - Pitch Rate

M - Pitching Moment

X, Z - Forces

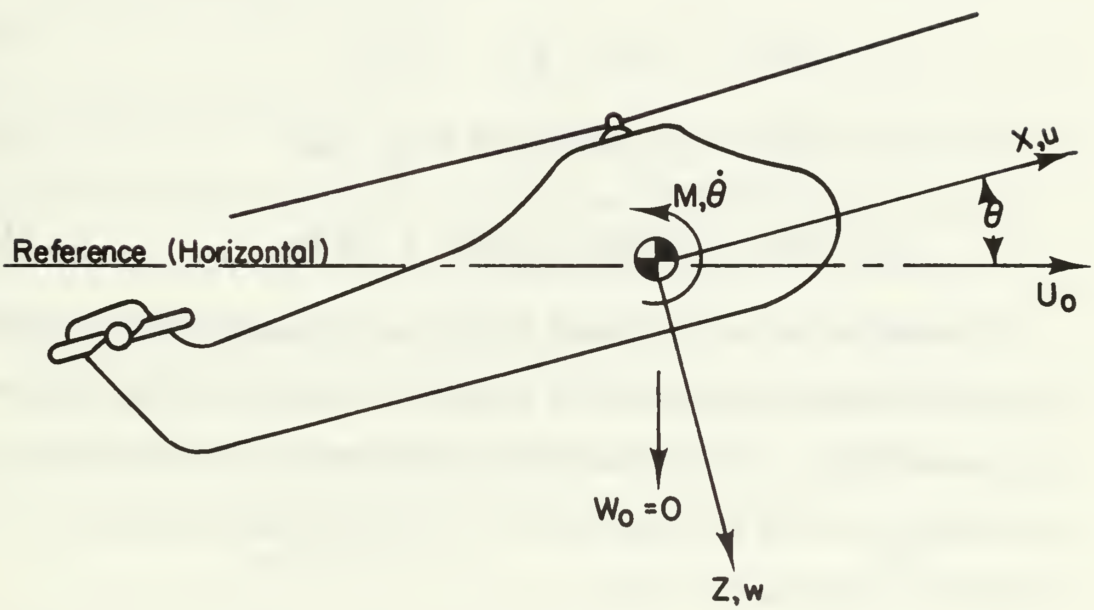

FIGURE 1. STABILITY AXIS REFERENCE SYSTEM

2. $M_{\dot{w}}=0$

3. Blade flapping neglected

4. Only cyclic inputs considered ($\delta_L=0$)

Assumption 1 was made to allow use of the stability axis reference system shown in Figure 1. Assumption 2 was based on values of $M_{\dot{w}}$ given in Refs. 5 and 7. The neglect of blade flapping introduced errors of unknown magnitude, but the assumption was considered acceptable for initial analysis. Neglecting the contributions of the collective control input was also not realistic, but was done for simplification of the problem. By incorporating these assumptions in the equations of motion, considerable simplification was achieved:

$$\dot{u} = X_u u + X_w w - g\theta + X_\delta \delta_\theta \tag{5}$$

$$\dot{w} = Z_u u + Z_w w + U_0 \dot{\theta} + Z_\delta \delta_\theta \tag{6}$$

$$\ddot{\theta} = M_u u + M_w w + M_q \dot{\theta} + M_\delta \delta_\theta . \tag{7}$$

An adaptive controller based on the above equations would require the measurement of u,w, and θ, and their derivatives plus cyclic displacement δ_θ. It was felt that measurement of the velocity perturbations would be undesirable. By introducing normal and tangential acceleration terms,

normal acceleration: $n = U_0 \dot{\theta} - \dot{w}$ (8)

tantential acceleration: $a = \dot{u}$, (9)

and substituting into Equations (5) and (6), expressions for u and w were found as functions of θ, a, n, and δ_θ. Substituting these for u and w in Equation (7), the pitching moment equation was

simplified to an expression free of dependence on the perturbation velocities, namely the relation

$$\ddot{\theta} = \bar{\beta}_q \dot{\theta} + \bar{\beta}_\theta \theta + \bar{\beta}_a a + \bar{\beta}_n n + \bar{\beta}_\delta \delta_\theta , \qquad (10)$$

where

$$\begin{aligned}
\bar{\beta}_q &= M_q \\
\bar{\beta}_\theta &= \frac{g(M_u Z_w - M_w Z_u)}{X_u Z_w - X_w Z_u} \\
\bar{\beta}_a &= \frac{\bar{\beta}_\theta}{g} \\
\bar{\beta}_n &= \frac{M_u X_w - M_w X_u}{X_u Z_w - X_w Z_u} \\
\bar{\beta}_\delta &= -\bar{\beta}_a X_\delta + \bar{\beta}_n Z_\delta + M_\delta .
\end{aligned} \qquad (11)$$

An adaptive controller based on Equation (10) requires measurement of $\ddot{\theta}$, $\dot{\theta}$, θ , a, n and δ_θ. It was presumed that these quantities would be measured to the required degree of accuracy. In order to keep the system as simple as possible, the effects of servos and actuators were neglected. The cyclic input was considered to be only the sum of pilot input and feedback input:

$$\delta_\theta = C + f. \qquad (12)$$

It was required to find a feedback function.

$$f = \Gamma_q \dot{\theta} + \Gamma_\theta \theta + \Gamma_a a + \Gamma_n n + \Gamma_\delta C , \qquad (13)$$

in which the gains Γ_q, Γ_θ , Γ_a, Γ_n and Γ_δ were variable over the range of flight conditions and made the aircraft behave as if

the variable parameters $\bar{\beta}_{q}$, $\bar{\beta}_{\theta}$, $\bar{\beta}_{a}$, $\bar{\beta}_{n}$ and $\bar{\beta}_{\delta}$ in Equation (10) were replaced by the fix parameters β_{q}, β_{θ}, β_{a}, β_{n} and β_{δ}. Substituting Equation (13) into Equation (10) yields

$$\ddot{\theta} \cong \beta_{q}\dot{\theta} + \beta_{\theta}\theta + \beta_{a}a + \beta_{n}n + \beta_{\delta}c \quad , \tag{14}$$

where

$$\begin{aligned}
\beta_{q} &= \bar{\beta}_{q} + \bar{\beta}_{\delta}\Gamma_{q} \\
\beta_{\theta} &= \bar{\beta}_{\theta} + \bar{\beta}_{\delta}\Gamma_{\theta} \\
\beta_{a} &= \bar{\beta}_{a} + \bar{\beta}_{\delta}\Gamma_{a} \\
\beta_{n} &= \bar{\beta}_{n} + \bar{\beta}_{\delta}\Gamma_{n} \\
\beta_{\delta} &= \bar{\beta}_{\delta}(1+\Gamma_{\delta}) .
\end{aligned} \tag{15}$$

Equation (14) would be an equality only when the variable gains assumed the proper values to satisfy the requirements of Equation (15). In order to determine the amount of error in Equation (14) the error function ϵ was defined as

$$\epsilon = \ddot{\theta} - \beta_{q}\dot{\theta} - \beta_{\theta}\theta - \beta_{a}a - \beta_{n}n - \beta_{\delta}C . \tag{16}$$

Substituting Equations (10) and (15) into (16) gives

$$\begin{aligned}
\epsilon = {} & (\bar{\beta}_{q} + \bar{\beta}_{\delta}\Gamma_{q} - \beta_{q})\dot{\theta} + (\bar{\beta}_{\theta} + \bar{\beta}_{\delta}\Gamma_{\theta} - \beta_{\theta})\theta \\
& + (\bar{\beta}_{a} + \bar{\beta}_{\delta}\Gamma_{a} - \beta_{a})a + (\bar{\beta}_{n} + \bar{\beta}_{\delta}\Gamma_{n} - \beta_{n})n + (\bar{\beta}_{\delta} + \bar{\beta}_{\delta}\Gamma_{\delta} - \beta_{\delta})c .
\end{aligned} \tag{17}$$

The values of the variable gains which would be required to drive the error to zero were found from Equation (17) to be

$$\Gamma_q^* = \frac{\beta_q - \bar{\beta}_q}{\bar{\beta}_\delta}$$

$$\Gamma_\theta^* = \frac{\beta_\theta - \bar{\beta}_\theta}{\bar{\beta}_\delta}$$

$$\Gamma_a^* = \frac{\beta_a - \bar{\beta}_a}{\bar{\beta}_\delta} \tag{18}$$

$$\Gamma_n^* = \frac{\beta_n - \bar{\beta}_n}{\bar{\beta}_\delta}$$

$$\Gamma_\delta^* = \frac{\beta_\delta}{\bar{\beta}_\delta} - 1 .$$

Define

$$B_q = \bar{\beta}_q + \bar{\beta}_\delta \Gamma_q - \beta_q$$

$$B_\theta = \bar{\beta}_\theta + \bar{\beta}_\delta \Gamma_\theta - \beta_\theta$$

$$B_a = \bar{\beta}_a + \bar{\beta}_\delta \Gamma_a - \beta_a \tag{19}$$

$$B_n = \bar{\beta}_n + \bar{\beta}_\delta \Gamma_n - \beta_n$$

$$B_\delta = \bar{\beta}_\delta + \bar{\beta}_\delta \Gamma_\delta - \beta_\delta .$$

Then

$$\epsilon = B_q \dot{\theta} + B_\theta \theta + B_a a + B_n n + B_\delta C . \tag{20}$$

Equation (20) would then approach zero only when B_q, B_θ, B_a, B_n and B_δ all approached zero simultaneously.

Following the method used in Ref. 1, let the non-negative error parameter V be defined by

$$2V = \frac{1}{K_q} B_q^2 + \frac{1}{K_\theta} B_\theta^2 + \frac{1}{K_a} B_a^2 + \frac{1}{K_n} B_n^2 + \frac{1}{K_\delta} B_\delta^2 , \tag{21}$$

where K_q, K_θ, K_a, K_n and K_δ are positive fixed numbers.

Differentiating Equation (21) with respect to time yields

$$\frac{dV}{dt} = \frac{1}{K_q} B_q \frac{dB_q}{dt} + \frac{1}{K_\theta} B_\theta \frac{dB_\theta}{dt} + \frac{1}{K_a} B_a \frac{dB_a}{dt} + \frac{1}{K_n} B_n \frac{dB_n}{dt} + \frac{1}{K_\delta} B_\delta \frac{dB_\delta}{dt}. \tag{22}$$

Differentiating Equation (19) with respect to time and substituting into Equation (22) we find

$$\frac{dV}{dt} = \bar{\beta}_\delta \left(\frac{1}{K_q} B_q \frac{d\Gamma_q}{dt} + \frac{1}{K_\theta} B_\theta \frac{d\Gamma_\theta}{dt} + \frac{1}{K_a} B_a \frac{d\Gamma_a}{dt} + \frac{1}{K_n} B_n \frac{d\Gamma_n}{dt} + \frac{1}{K_\delta} B_\delta \frac{d\Gamma_\delta}{dt} \right). \tag{23}$$

If we are to have B_q, B_θ, B_a, B_n and B_δ approaching zero, $\frac{dV}{dt}$ must be negative, by virtue of Equation (21). By choosing

$$\begin{aligned}
\frac{d\Gamma_q}{dt} &= \pm K_q \dot{\theta} G \\
\frac{d\Gamma_\theta}{dt} &= \pm K_\theta \theta G \\
\frac{d\Gamma_a}{dt} &= \pm K_a a G \\
\frac{d\Gamma_n}{dt} &= \pm K_n n G \\
\frac{d\Gamma_\delta}{dt} &= \pm K_\delta C G,
\end{aligned} \tag{24}$$

Equation (23) would be written

$$\frac{dV}{dt} = \bar{\beta}_\delta (\pm \epsilon) G. \tag{25}$$

The sign of Equations (24) and (25) was determined partly by the sign of $\bar{\beta}_\delta$ which is dependent on the stability derivatives as given in Equation (11). For the test model used, $\bar{\beta}_\delta$ was positive so that by choosing the negative sign Equation (25) became

$$\frac{dV}{dt} = -\bar{\beta}_\delta \, \epsilon G, \tag{26}$$

Then following the method presented in Ref. 1, let

$$G = Sgn\,\epsilon = \begin{cases} +1 & \epsilon > \epsilon_o \\ 0 & -\epsilon_o \le \epsilon \le \epsilon_o \\ -1 & \epsilon < -\epsilon_o \end{cases} , \qquad (27)$$

so that

$$\frac{dV}{dt} = -\bar{\beta}_s \, |\epsilon| , \qquad (28)$$

thus assuring that $\frac{dV}{dt}$ would always be negative, and that B_q, B_θ, B_a, B_n and B_δ could approach zero.

Expressions for the variable gains were found from Equation (24) as

$$\begin{aligned} \Gamma_q &= -\int K_q \dot{\theta} \, Sgn\,\epsilon \, dt \\ \Gamma_\theta &= -\int K_\theta \theta \, Sgn\,\epsilon \, dt \\ \Gamma_a &= -\int K_a a \, Sgn\,\epsilon \, dt \\ \Gamma_n &= -\int K_n n \, Sgn\,\epsilon \, dt \\ \Gamma_\delta &= -\int K_\delta C \, Sgn\,\epsilon \, dt . \end{aligned} \qquad (29)$$

A block diagram of the system as derived is given in Figure 2.

Although it was guaranteed that $\frac{dV}{dt}$ would always be negative, with V approaching zero, the stability of the system was not insured for reasons given in Ref. 2: mainly that the manner in which the gains converged toward ideal values in the steady state was a function of the input command. In order to determine whether the system would actually stabilize the model, and produce the desired handling qualities, it was necessary to continue the evaluation by use of the analog computer.

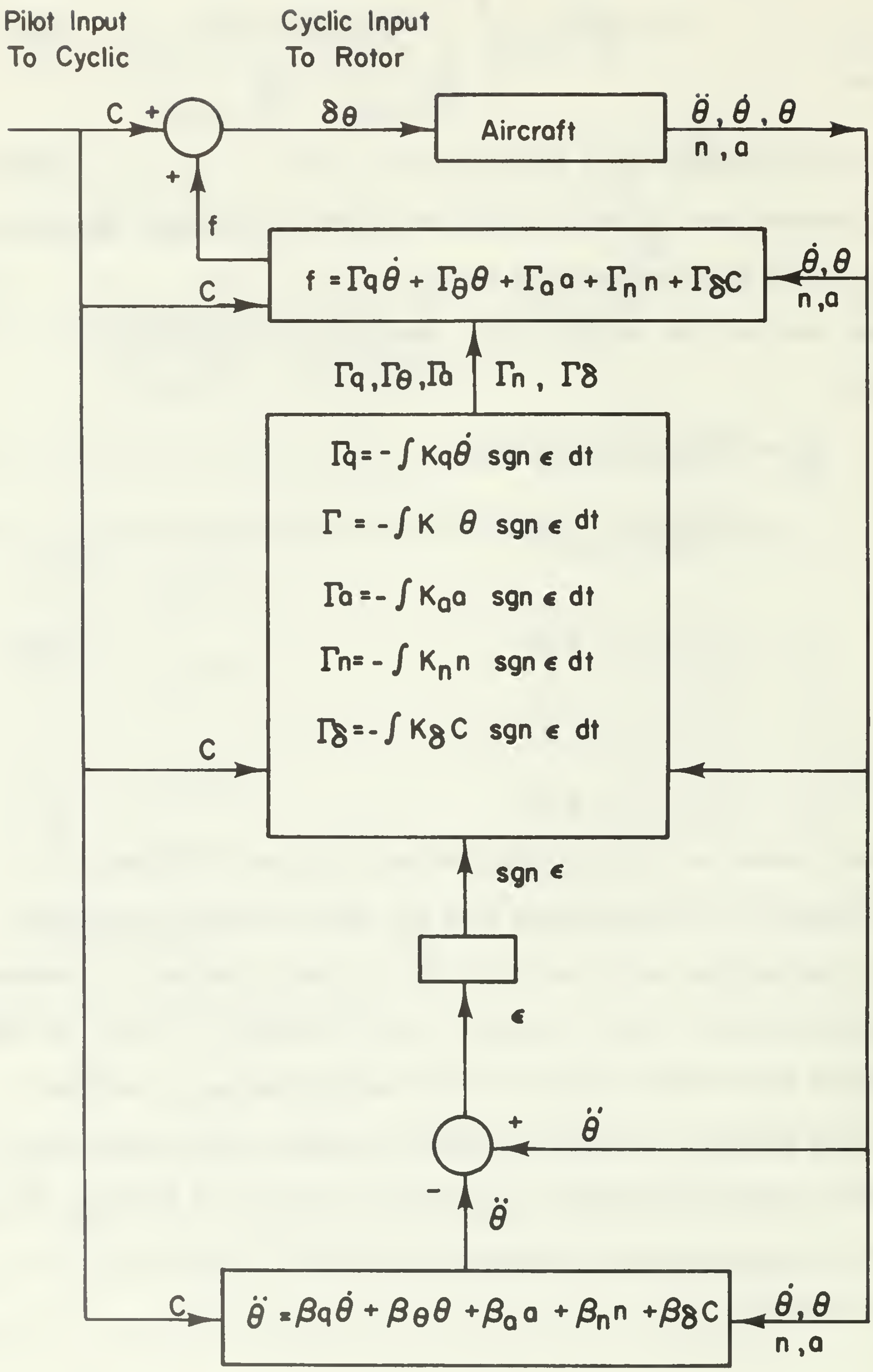

FIGURE 2. HELICOPTER LONGITUDINAL ADAPTIVE CONTROL SYSTEM BLOCK DIAGRAM

B. MODIFICATION OF THE C^*- CRITERION

The C^*-criterion, originally developed by the Boeing Company, is used to determine whether the handling qualities of the aircraft are within desired limits. The criterion states that the time-response curve of the quantity C^*, for an abrupt input command, must fall within a certain envelope. The envelope used for fixed-wing aircraft is shown in Ref. 2, but it cannot be assumed that this envelope would be acceptable for helicopters. Rang [2] conjectured that the C^*-criterion should not be met directly but rather achieved by letting the coefficients of the C^* -expression be taken in the range of the coeffecients of the corresponding short period equation. This method will allow for the variation in handling qualities required over the range of the flight envelope, especially at low-q conditions where helicopters normally operate.

For fixed-wing aircraft the quantity C^* is taken to be

$$C^* = n + \ell\ddot{\theta} + U_c\dot{\theta} \, , \tag{30}$$

where ℓ is the distance of the pilot forward of the center-of-gravity and U_c is a fixed number called the cross-over velocity. Equation (10) introduced the two additional terms of pitch attitude and tangential acceleration for the present study. Incorporating these changes, Equation (30) was modified to

$$C^* = n + a + \ell\ddot{\theta} + U_c\dot{\theta} + V_c\theta \, . \tag{31}$$

The closeness of the center-of-gravity to the pilot, and the relatively low performance demands, would make the pitch acceleration, normal acceleration and tangential acceleration have a small effect on C^*. In helicopters, the pitch rate and particularly the pitch attitude are the quantities that normally have to be controlled.

By requiring that C^* be a multiple of the input command

$$C^* = KC, \tag{32}$$

Equation (31) could be rewritten as

$$\ddot{\theta} = -\frac{u_c}{I}\dot{\theta} - \frac{V_c}{I}\theta - \frac{1}{I}n - \frac{1}{I}a - \frac{K}{I}C, \tag{33}$$

which is identical to the form of Equation (10). Requiring the time response of C^* to fall within the prescribed envelope is equivalent to requiring the system to hold the coefficients in Equation (14) constant. Reference 2 indicates that these β-parameters should be chosen within the range of the $\bar{\beta}$-parameters. Having determined the desired envelope for the C^*-response, it would be hoped that the handling qualities of the helicopter could be shaped as desired by variation of the β-parameters without dependence on an outside loop. It would be desirable to let β_a equal zero in order to completely eliminate the effect of the tangential acceleration on C^* and to simplify Equation (31) by adding only one additional term.

C. ANALOG SIMULATION OF THE ADAPTIVE CONTROLLER

Evaluation of the controller was made on an all-analog system. Initial evaluation, consisting of determining the effect of the various β-parameters on the system and finding values which would stabilize the model and produce desirable handling qualities, was completed on the EAI 580 analog computer operated by the Department of Aeronautics at the Naval Postgraduate School. Final evaluation, showing that the system would drive the error function to zero as predicted, and finding satisfactory values for the K-parameters, was done on the Comcor CI 5000 analog computer operated by the Department of Electrical Engineering at the Naval Postgraduate School. Diagrams

of the analog computer circuits used are given in figures 3-5. The potentiometer settings are listed in Table III at the end of the Chapter.

The OH-5 helicopter was chosen as a test model because of the availability of data on its stability derivatives. These, and the computed values of $\bar{\beta}_q$, $\bar{\beta}_\theta$, $\bar{\beta}_a$, $\bar{\beta}_n$ and $\bar{\beta}_\delta$ are listed in Table I at the end of the Chapter. All values for the stability derivatives were taken from Ref. 7 with the exception of X_δ. Initial computations using the values given in Ref. 7 produced a value for $\bar{\beta}_\delta$ in the hover condition which was very near zero, and negative in sign. Referring to Equation (18), it was seen that the small value would cause the variable gains to approach very large values and the change in sign would probably produce an unstable system. Since the purpose of the tests was to evaluate the system, not a particular model, the value of X_δ was changed from -10.41 to -5.00 in order to bring $\bar{\beta}_\delta$ up to a large enough positive value to keep the variable gains at a reasonable level. This change did not alter the responses of the free aircraft in any essential manner. It is not known if the sign of $\bar{\beta}_\delta$ usually changes sign, if this was an unusual happening, or if the data were incorrect.

In order to test the full range of flight conditions, the evaluations were conducted at hover, 40 mph and 140 mph. It was required to find values for β_q, β_θ, β_a, β_n and β_δ which would stabilize the model, require values for the variable gains which were not too large and still provide desirable handling qualities. It was beyond the scope of these initial evaluations to determine the proper envelope for the C*-criterion. It was decided to use the guidelines as set forth in Ref. 5 with regard to handling qualities. It was also desired to keep β_q, β_θ, β_a, β_n and β_δ within the range of $\bar{\beta}_q$, $\bar{\beta}_\theta$, $\bar{\beta}_a$, $\bar{\beta}_n$ and $\bar{\beta}_\delta$ if possible. Plots of β vs Γ^* given

in Figures 6-10 at the end of Chapter III showed that large variations in the β-parameters produced very little change in the steady-state variable gains at 140 mph but gave very large changes at hover and 40 mph. It was therefore suspected that, if values of the β-parameters could be found which satisfied the lower speed flight conditions, very little modification would be necessary to satisfy the high speed condition. Although the C^*-criterion was not considered, it was hoped that the requirements could be met with β_a set equal to zero and that the acceptable handling qualities would be produced by varying the other β-parameters.

The final values of β_q, β_θ, β_a, β_n and β_δ used, and the corresponding values of the ideal steady state gains are given in Table II and presented graphically in Figures 6-10. The values of the fixed gains K_q, K_θ, K_a, K_n and K were obtained by a trial and error method once the final values of β_q, β_θ, β_a, β_n and β_δ were determined. The results of the evaluations are presented in Chapter III.

TABLE I

STABILITY DERIVATIVES AND ASSOCIATED

VALUES FOR THE OH-5 HELICOPTER

QUANTITY	HOVER	40 MPH	140 MPH
X_u	-0.0058	-0.0166	-0.0780
X_w	0	0.0120	0.1480
X_δ^*	-5.000	-3.600	14.00
Z_u	0	0.0200	0.1360
Z_w	-0.8980	-0.8980	-0.8980
Z_δ	0	-53.00	-184.0
M_u	0.0015	0.0030	0.0097
M_w	-0.0076	-0.0215	-0.0696
M_q	-0.0070	-0.1800	-0.6300
M_δ	2.680	0.970	3.530
$\bar{\beta}_q$	-0.0070	-0.1800	-0.6300
$\bar{\beta}_\theta$	-4.830	-4.060	-.6760
$\bar{\beta}_a$	-0.1500	-0.1260	0.0210
$\bar{\beta}_n$	-0.0085	-0.0217	-0.0800
$\bar{\beta}_\delta$	1.960	1.650	18.00

*adjusted from an actual value of -10.41

TABLE II

VALUES AND ASSOCIATED IDEAL STEADY STATE GAINS

QUANTITY	HOVER	40 MPH	140 MPH
β_g	-1.50	-1.50	-1.50
β_θ	-1.288	-1.288	-1.288
β_a	0	0	0
β_n	-0.080	-0.080	-0.080
β_δ	18.00	18.00	18.00
Γ_g^*	-0.762	-0.868	-0.048
Γ_θ^*	1.807	1.824	-0.109
Γ_a^*	0.0765	0.0829	0
Γ_n^*	-0.0365	-0.0384	0
Γ_δ^*	8.184	10.842	0

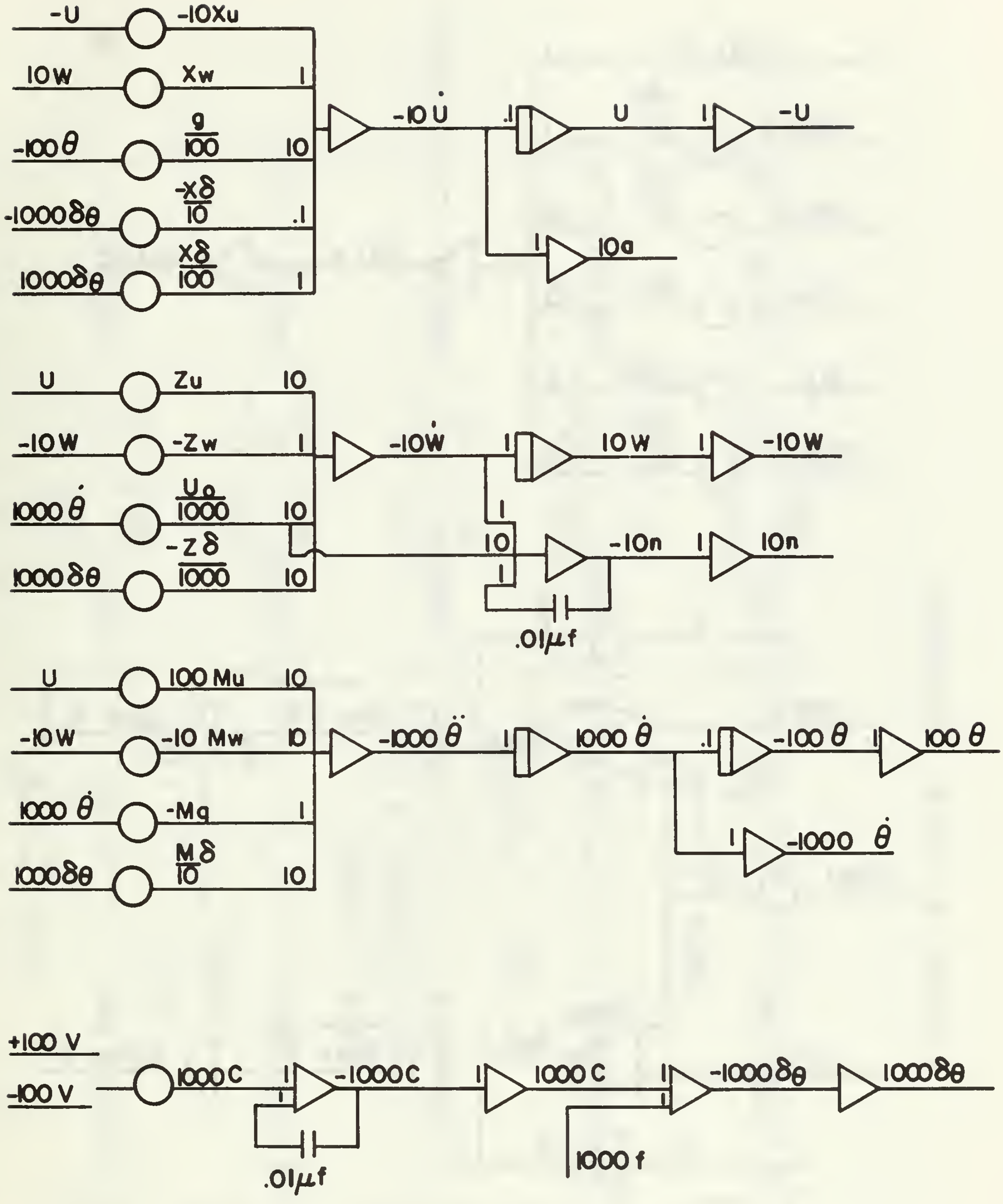

FIGURE 3. LONGITUDINAL AIRCRAFT EQUATIONS OF MOTIONS

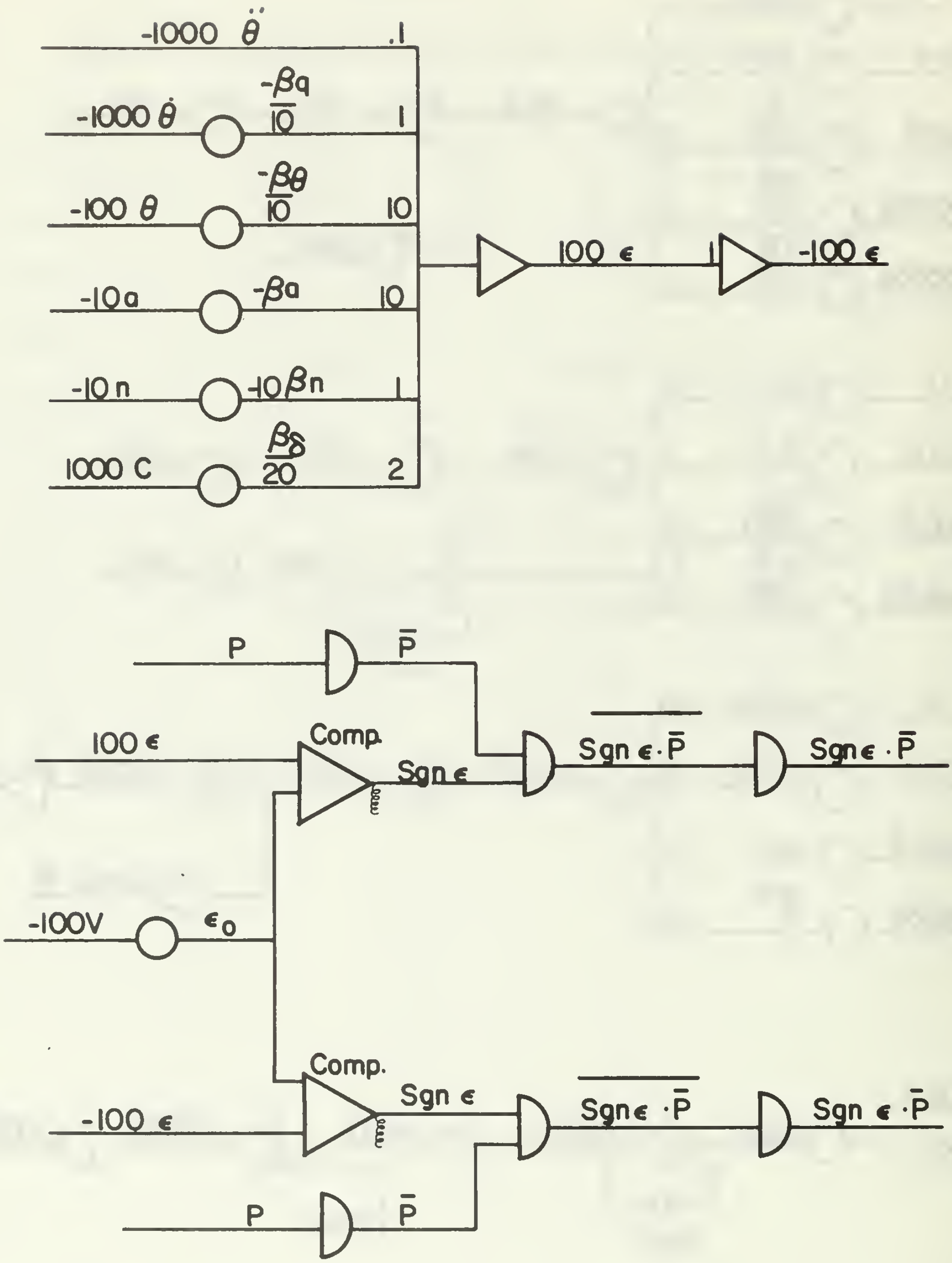

FIGURE 4. ERROR AND Sgn ERROR

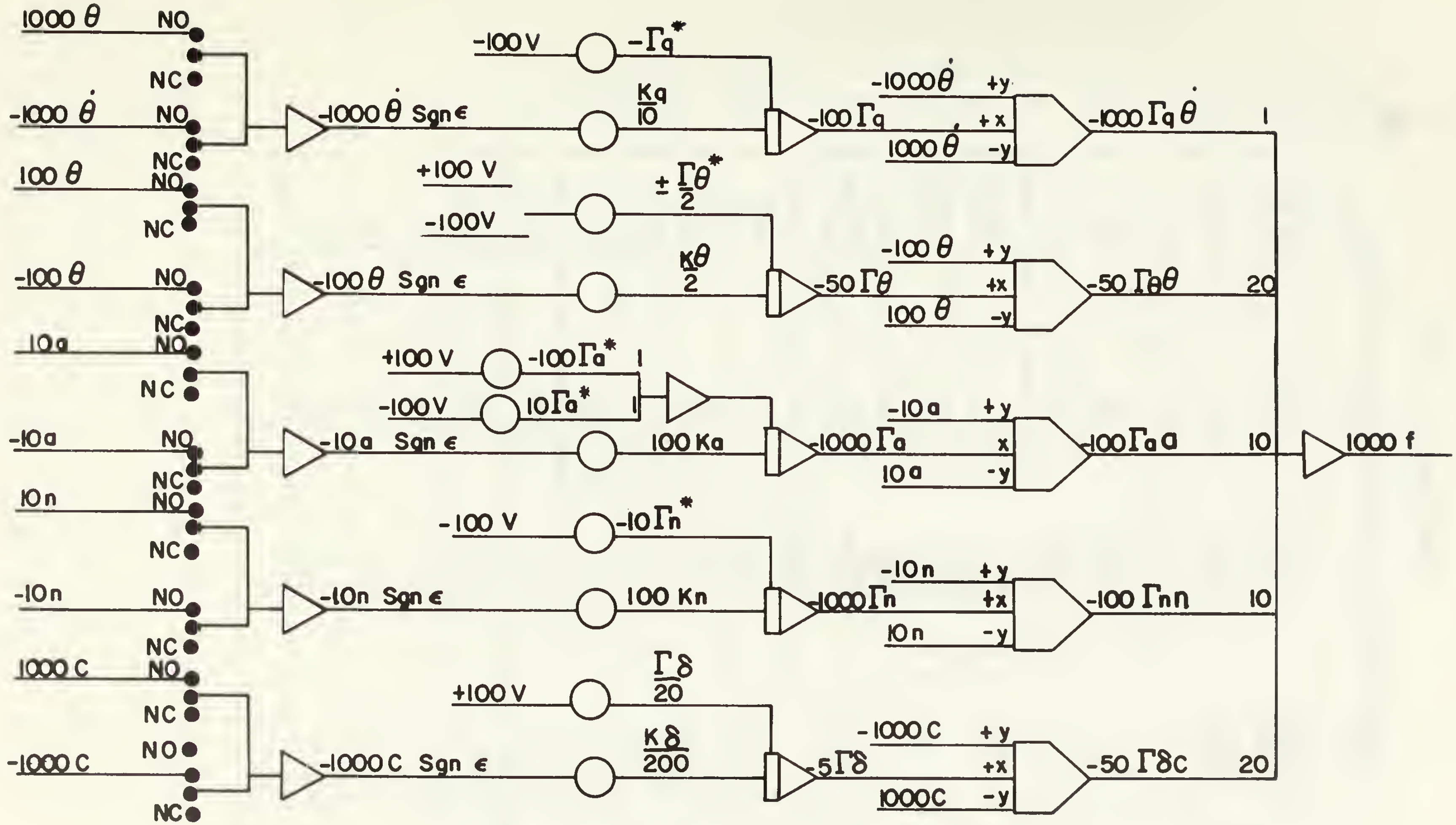

FIGURE 5. CYCLIC FEEDBACK

TABLE III

ANALOG COMPUTER PONTENTIOMETER SETTINGS

POT	QUANTITY	HOVER	40 MPH	140 MPH
00	$-10X_u$	.058	.166	.780
01	X_w	0	.012	.148
02	$g/100$	.322	.322	.322
03	$-X_\delta/10$	.500	.360	0
04	$X\ /100$	0	0	.140
05	Z_u	0	.020	.136
06	$-Z_w$	.898	.898	.898
07	$U_o/1000$	0	.058	.206
10	$-Z_\delta/1000$	0	.053	.184
11	$100M_u$	.150	.300	.970
12	$-10M_w$	.076	.215	.696
13	$-M_q$	.007	.180	.630
14	$M_\delta/10$	.268	.097	.353
15	$1000C$	.050	.050	.050
16	100ϵ	.002	.002	.002
17	$-\beta_\delta/10$	.150	.150	.150
20	$-\beta_\theta/10$	.129	.129	.129
21	$-\beta a$	0	0	0

TABLE III continued

POT	QUANTITY	HOVER	40 MPH	140 MPH
22	$-10\beta_n$	.800	.800	.800
23	$\beta_\delta/20$	.900	.900	.900
	Pots 24-30 used for actual controller			
24	$K_q/10$	.400	.400	.400
25	$K_\theta/10$	.500	.500	.500
26	$1000K_a$	.500	.500	.500
27	$1000K_n$	.500	.500	.500
30	$K_\delta/2000$	.150	.150	.150
	Pots 31-36 used for ideal controller			
31	$-\Gamma_\delta^*$	.762	.868	.048
32	$\pm\Gamma_\theta^*/2$	+.912	+.912	-.055
33	$-100\Gamma_a^*$	0	0	.117
34	$10\Gamma_a^*$	.765	.830	0
35	$-10\Gamma_n^*$	.360	.380	0
36	$-\Gamma_\delta^*/20$	.409	.542	0

III. RESULTS

Evaluation of the system on the analog computer showed that the model, which was unstable in the free condition, could be easily stabilized through proper choice of the β-parameters. Producing the desired handling qualities based on Ref. 5 proved to be very difficult however, and a good combination was not found.

Under actual operation, with the controller computing the variable gains required to drive the responses toward the previously determined ideal responses, the controller did behave as predicted by driving the error function to zero in a very short time period.

The graphical results of the analog simulation are given in Figures 11 thru 21. All traces were run at 5 mm per second, with the volts per line (V/L) scale varied as shown on the graphs. A step input of -5.0 volts, corresponding to a slight forward deflection of the cyclic control, was used for all tests. As stated earlier, collective pitch inputs were not considered.

The model was found to be very unstable in the hover condition, with the stability increasing as speed increased. Responses of the free aircraft at the three flight conditions are shown in Figure 11. At hover the step input caused large oscillating responses which diverged rapidly. The aircraft was marginally stable at 40 mph producing oscillations which took a considerable length of time to die out. Only small oscillations, which were eliminated very quickly, were evident at 140 mph.

Selection of the best combination of β-parameters required investigation of the effect of each parameter individually. Figures 6

thru 10 show the relationship of the values of $\bar{\beta}$ at each flight condition and the final value of β which was used. It was found that at all flight conditions the stability could be increased by decreasing the value of β_γ, β_θ and β_n below $\bar{\beta}_\gamma$, $\bar{\beta}_\theta$ and $\bar{\beta}_n$, respectively. Conversely, the values of β above $\bar{\beta}$ at any flight condition would decrease the stability. By placing β within the range of $\bar{\beta}$, one or more flight conditions could be stabilized and one or more destabilized. It was required that β_a be greater than $\bar{\beta}_a$ for increased stability. β_δ affected only the magnitude of the responses and not the stability. Increasing the value of β_δ greatly increased the magnitude of the responses, while decreasing β_δ below a certain point could actually cause control reversal.

Values which increased stability also greatly reduced the magnitude of the responses and slowed the response time by a large degree. Decreasing the stability increased the magnitude of the responses and speeded up the response time. Therefore, it became necessary to choose a combination of β-parameters which would provide the necessary amount of stability without slowing the reaction time. By attempting to keep the values of β within the range of $\bar{\beta}$, each value of β would have to produce a different effect at the various airspeeds, i.e., stabilizing at some and destabilizing at others.

As shown in Figure 6, β_γ was the only parameter which was outside the range of $\bar{\beta}$. This was required in order to provide increased stability at all flight conditions. β_θ was used to increase the speed of response both at hover and at 40 mph, and to increase the stability at 140 mph. Figure 7 indicates the position of β_θ relative to $\bar{\beta}_\theta$. β was set at zero as desired,

as seen in Figure 8, resulting in greatly increased stability at hover and at 40 mph and in slightly decreased stability at 140 mph. Letting β_n equal $\bar{\beta}_n$ at 140 mph produced no effects at high speed while adding stability at 40 mph. Neglecting the collective input caused elimination of normal acceleration at hover so that β_n had no effect in that condition. The overall effect of the above settings added stability to all flight conditions but also greatly reduced the magnitude of responses. In order to increase the magnitudes it was necessary to let β_δ assume a large value, as indicated in Figure 10.

The resulting system produced responses which were adequately stable with acceptable magnitudes of response. The increase in stability gained, however, slowed the speed of response more than desired. Attempts to increase the speed of response in order to meet the requirements of Ref. 5 resulted in drastic decreases in the stability at all flight conditions.

Initial evaluation, using the listed β-parameters, produced a very high frequency oscillation in normal acceleration. This oscillation was traced to a phase shift in the amplifiers located in the normal acceleration loop. Use of a small feedback capacitor as shown in Figure 3 eliminated the problem.

Figures 13, 16 and 19 indicate the free, ideal and actual responses at the various flight conditions. The ideal responses were obtained by setting the fixed-gain K-parameters at zero, and by setting the ideal, steady-state gains listed in Table II as initial conditions (shown in Figure 5). The actual responses were obtained by letting the system compute the required variable gains. The amount by which the actual responses varied from the ideal was

indicated by the error ϵ . It should be noted that all plots were obtained by introducing the step input via the reset position on the computer. Under normal operation the constant movement of the cyclic control would cause the controller to drive the variable gains to steady-state values, which would theoretically remain unchanged at a given flight condition. With these gains at a constant setting, the time required to drive the error to zero would be greatly reduced. Using only the initial step input, a noticeable error was seen at hover and 40 mph, which reduced to zero within three seconds. The error at 140 mph was much less and was reduced to zero in less than one second.

The errors recorded were actually variations in the pitch acceleration as determined by the combination of pitch rate and attitude, normal and tangential acceleration, and the cyclic input. Examination of Figure 13 indicates that at hover only the pitch rate varied noticeably from ideal, while the other responses were essentially ideal from the moment of command input. Figure 16 shows that both pitch rate and normal acceleration varied from ideal, while pitch attitude and tangential acceleration were as desired. As predicted by the low error produced, all actual responses at 140 mph were very close to ideal, as shown in Figure 19.

Figures 15, 18 and 21 give a comparison between ideal and actual cyclic inputs as determined by the sum of the constant step input and the feedback input. These variations were limited to the first few seconds of operation and corresponded to the error signal received.

The variable gains were found to be considerably different from the ideal gains calculated to produce ideal responses. Figures 14, 17 and 19 show that the actual gains computed by the system were much smaller than the ideal gains and, in some cases are even opposite

in sign. From the discrepancy in ideal and actual gains, it appeared that the feedback required to drive the system error to zero was only a weak function of the variable gains computed.

This weak dependence made the selection of values for the fixed K-parameters relatively easy. By letting K_q=4, K_θ=1, K_a=K_n=0.005 and K =300, the responses shown in Figures 11 thru 21 were produced. A fairly wide variation of these parameters changed the values of the variable gains computed, but did not effect the nature of the responses in a visible manner. Only K_δ appeared to be critical. Lowering K_δ below 300 caused considerably greater error, which took longer to reach zero.

In summary, the analog computer evaluations showed that the controller would stabilize the system in the manner predicted, but did not offer the range of handling qualities desired.

The most probable reason for the poor handling qualities appeared to be the wide variation in the stability of the free aircraft over the range of flight conditions. Figures 6-10 show that the β-parameters at both hover and at 40 mph are fairly close, with the paramaters at 140 mph being a relatively great distance away. This wide variation enabled the shaping of desirable handling qualities at hover and at 40 mph, or at 140 mph, but not at all three conditions. It is not known if this characteristic is common to all helicopters.

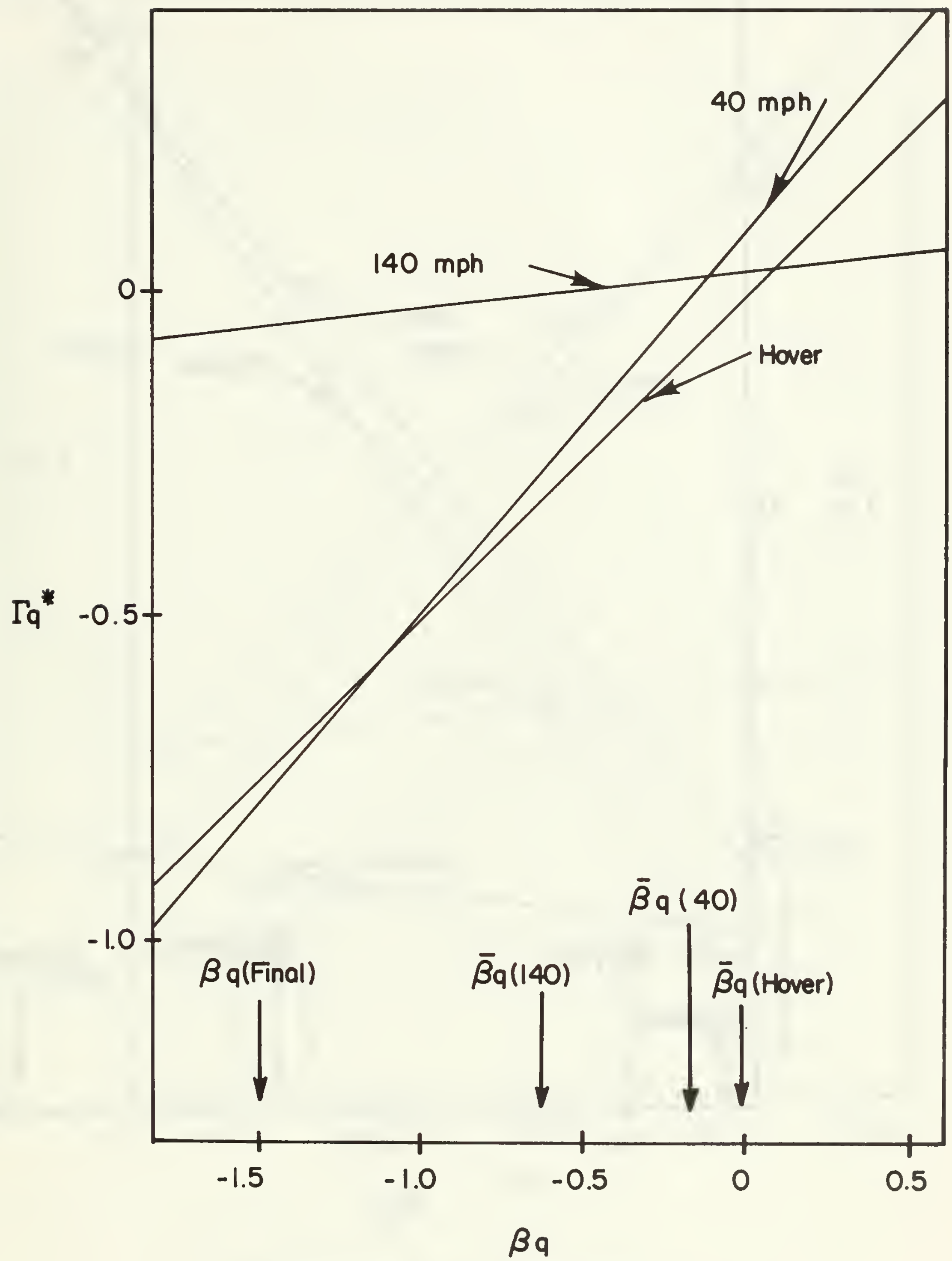

FIGURE 6. β_q VS. Γ_q^*

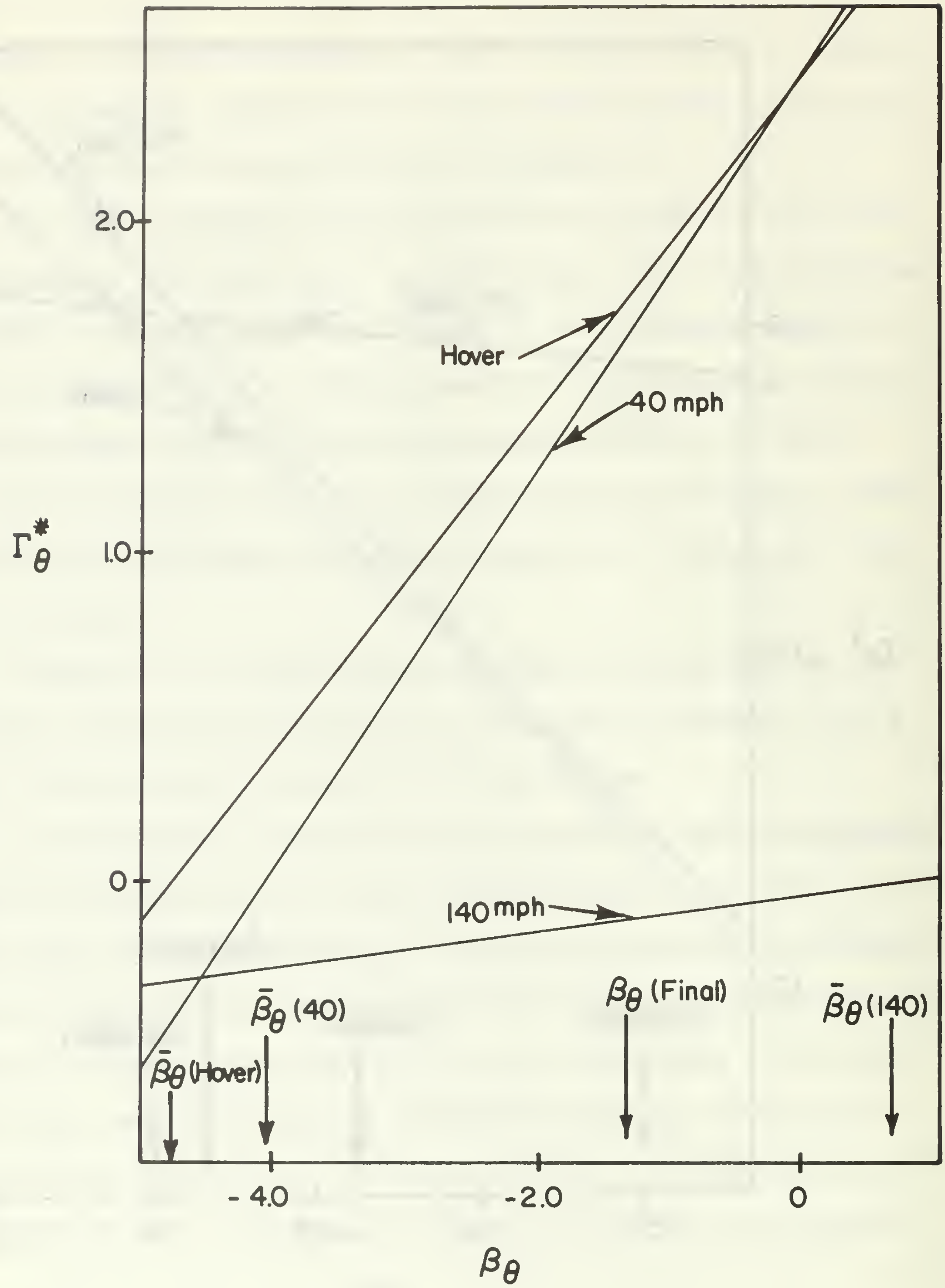

FIGURE 7. β_θ VS. Γ_θ^*

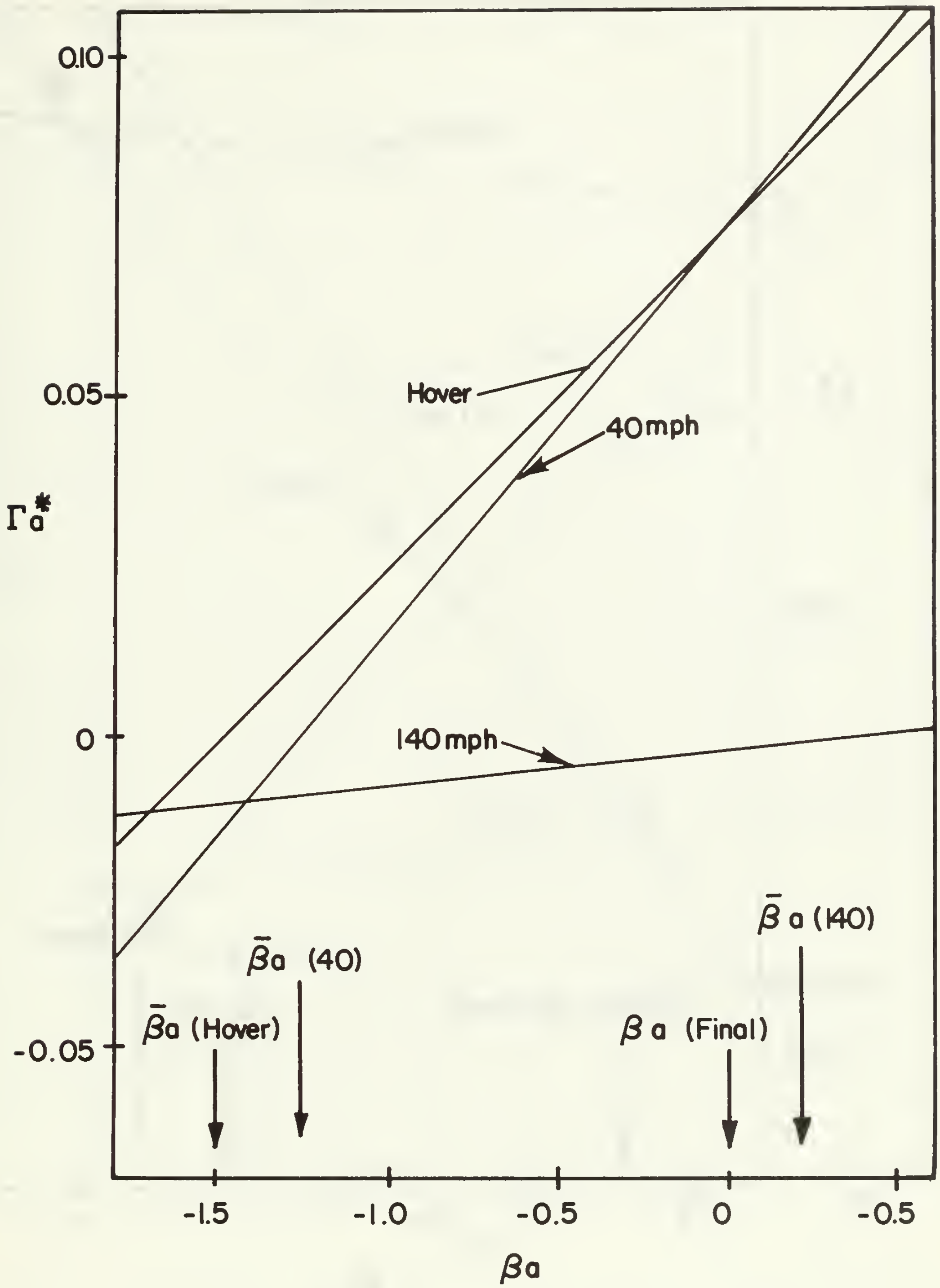

FIGURE 8. βa VS. Γa

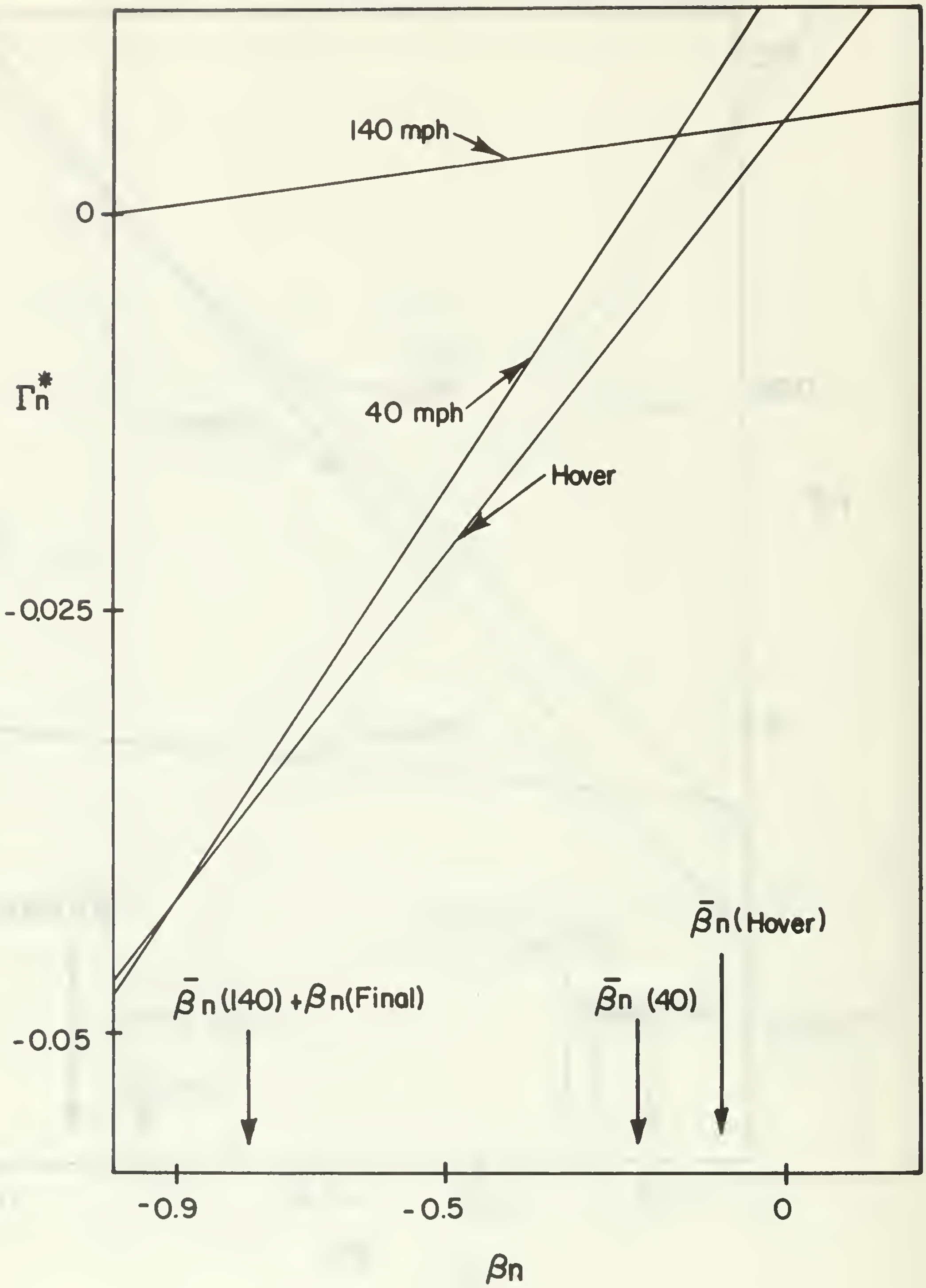

FIGURE 9. β_n VS. Γ_n^*

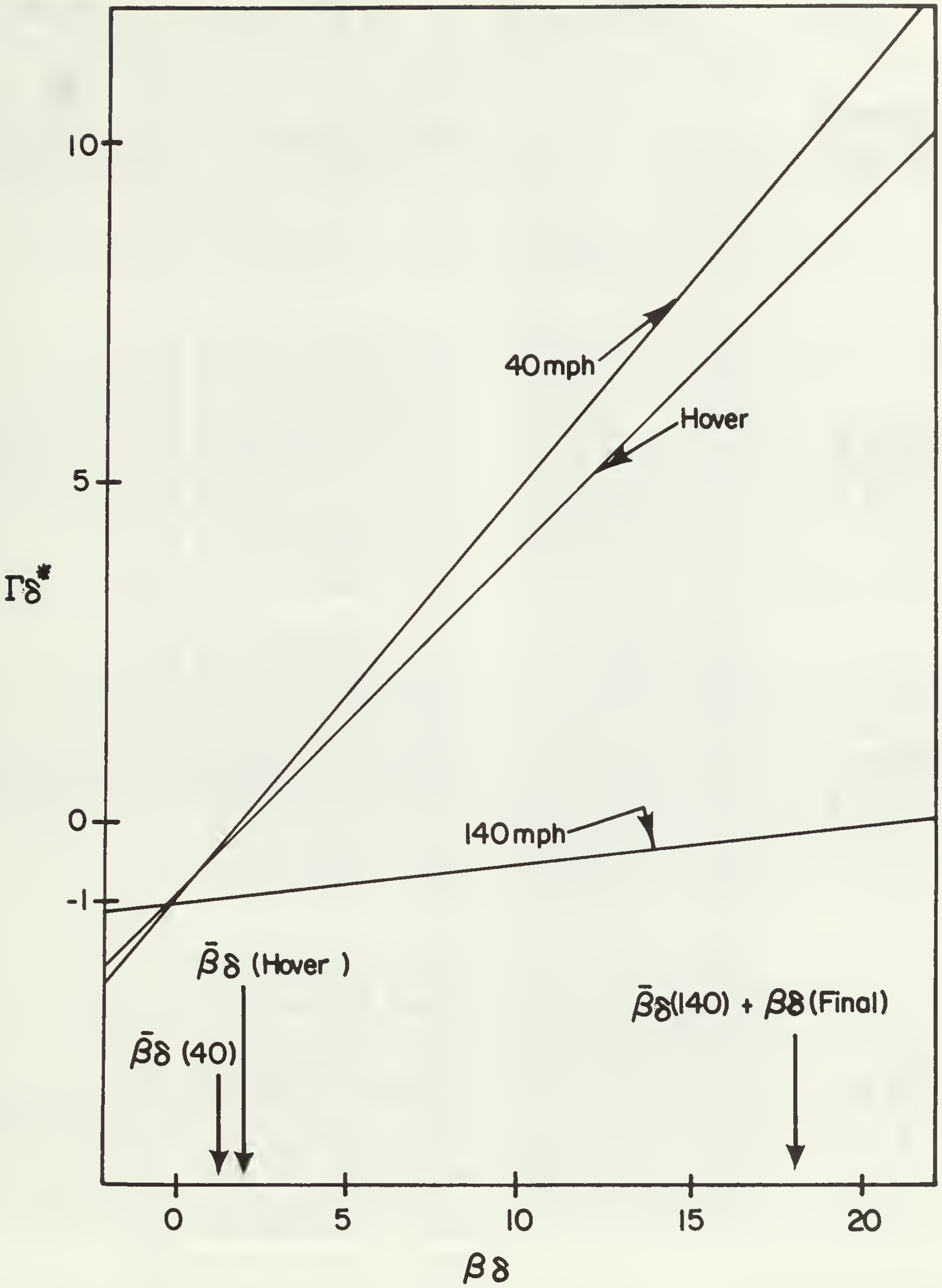

FIGURE 10. $\beta\delta$ VS. $\Gamma\delta^*$

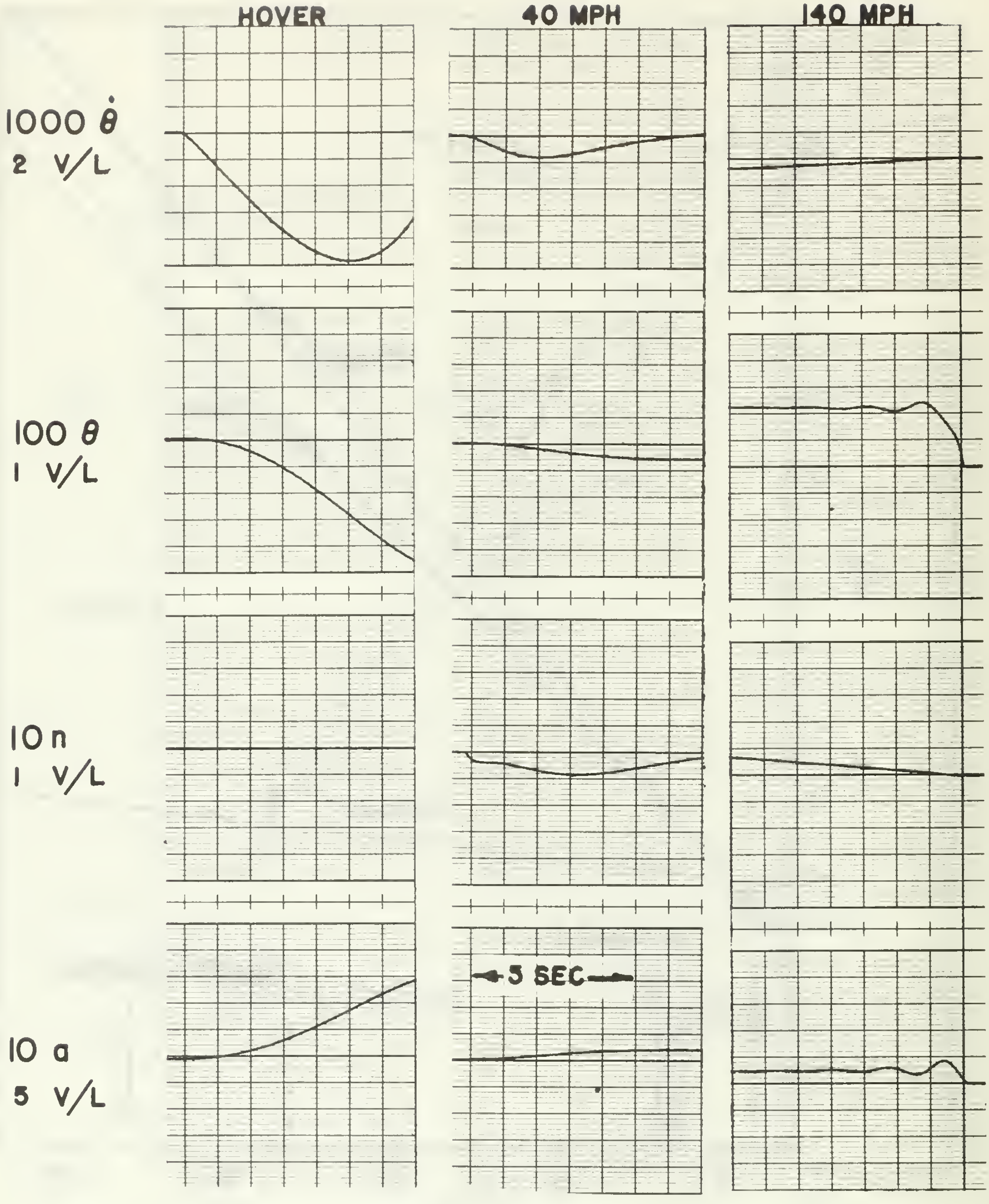

FIGURE 11. FREE AIRCRAFT

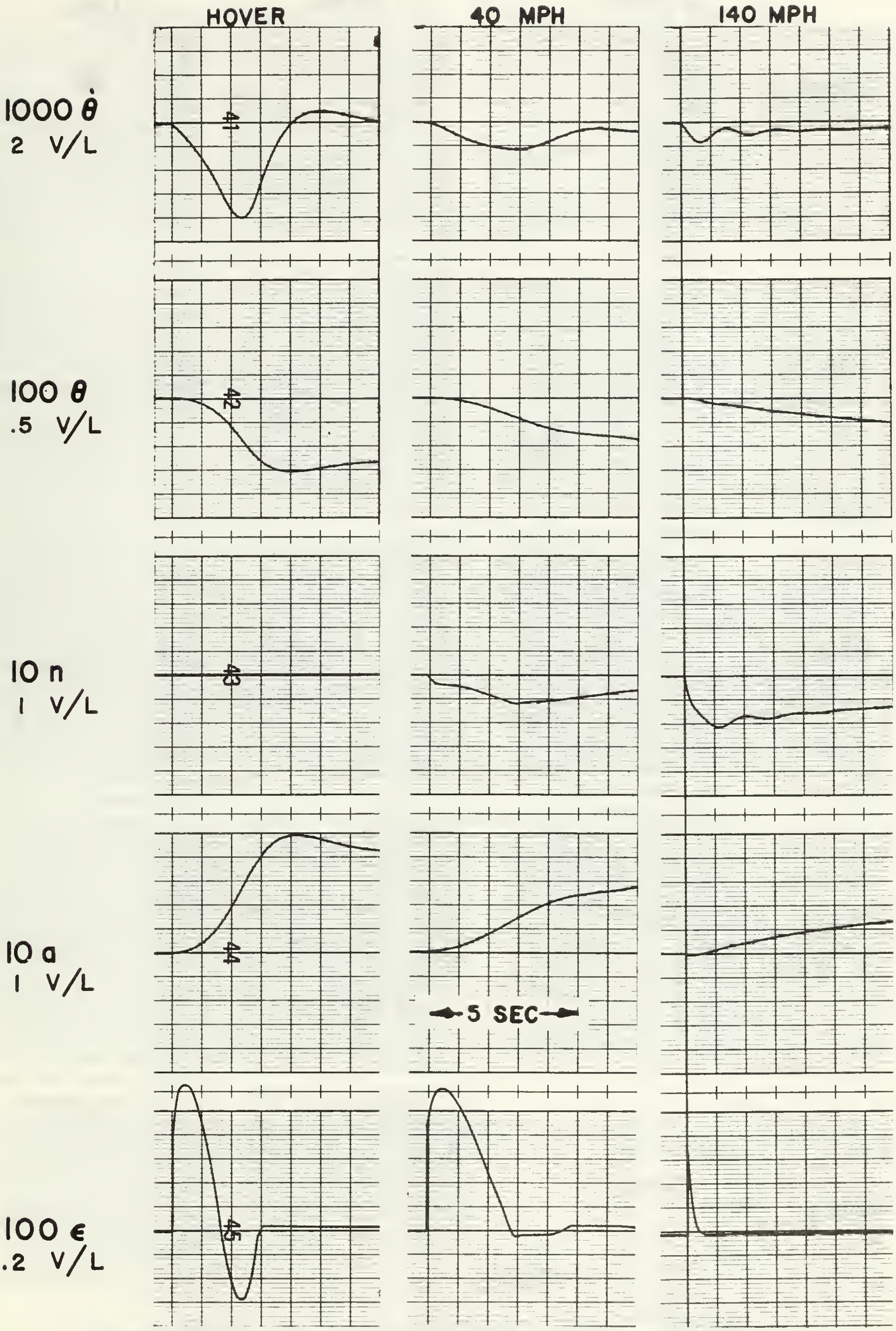

FIGURE 12. AIRCRAFT WITH ADAPTIVE CONTROL

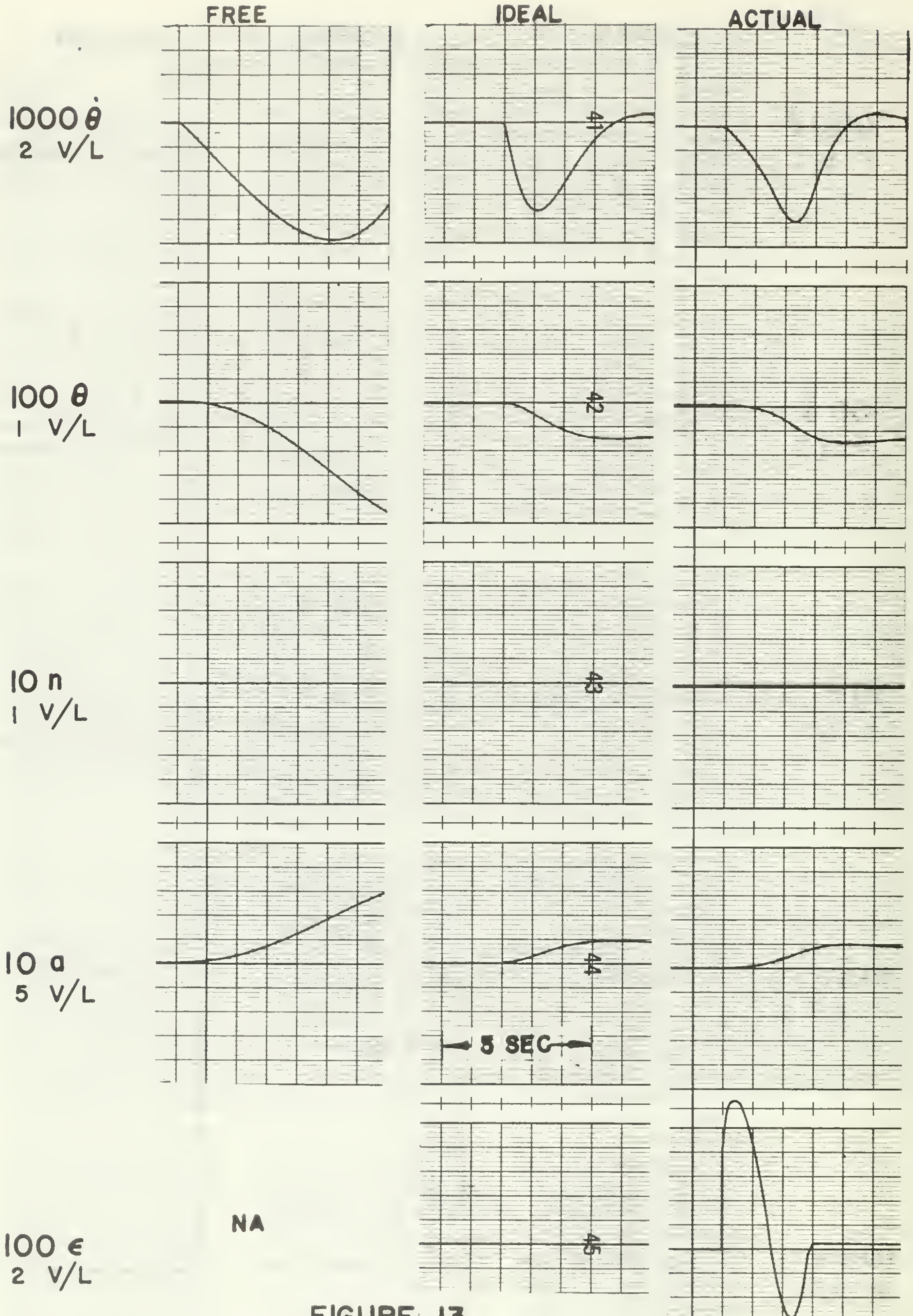

FIGURE 13.

EFFECT OF CONTROLLER AT HOVER

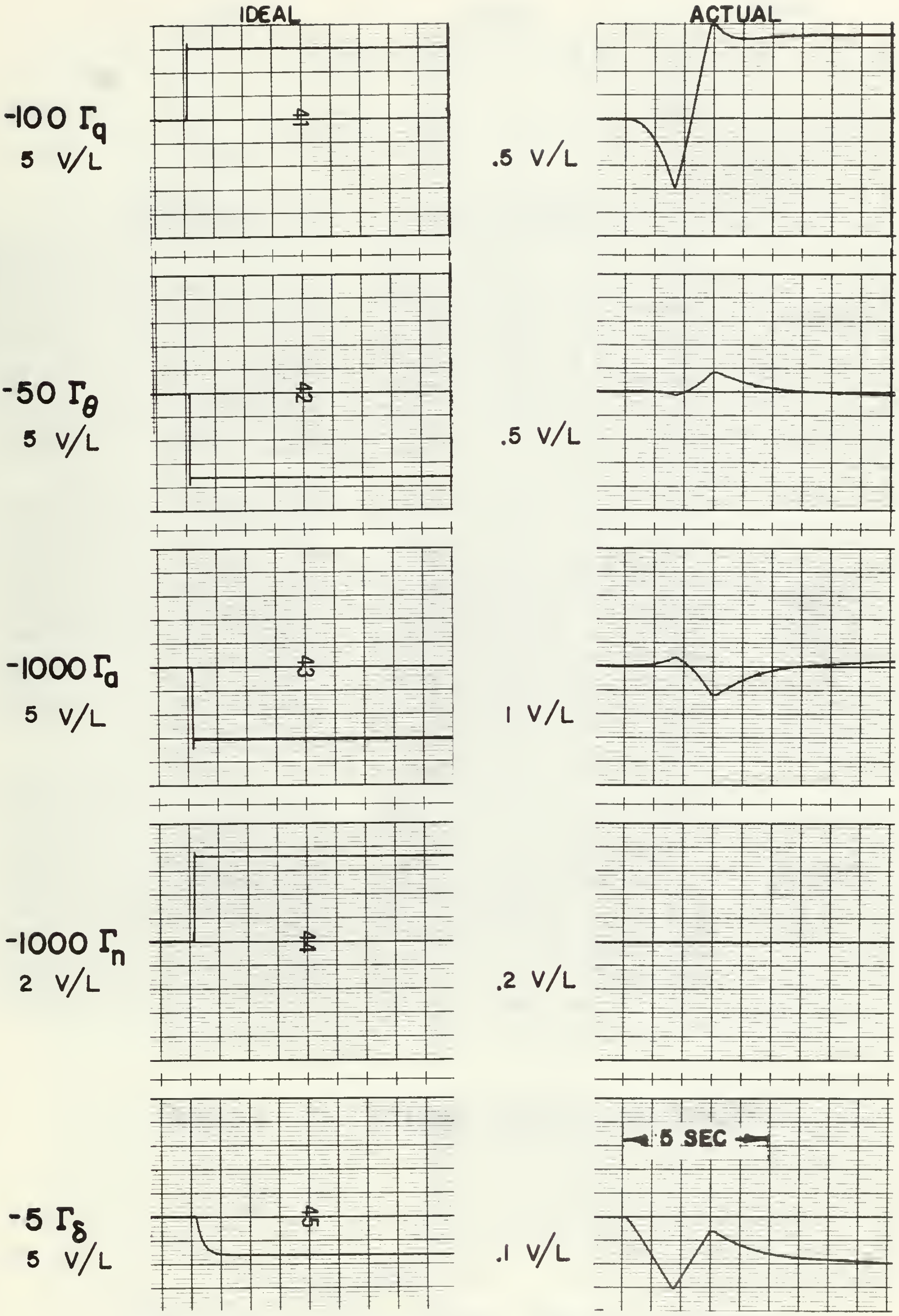

FIGURE 14. VARIABLE GAINS AT HOVER

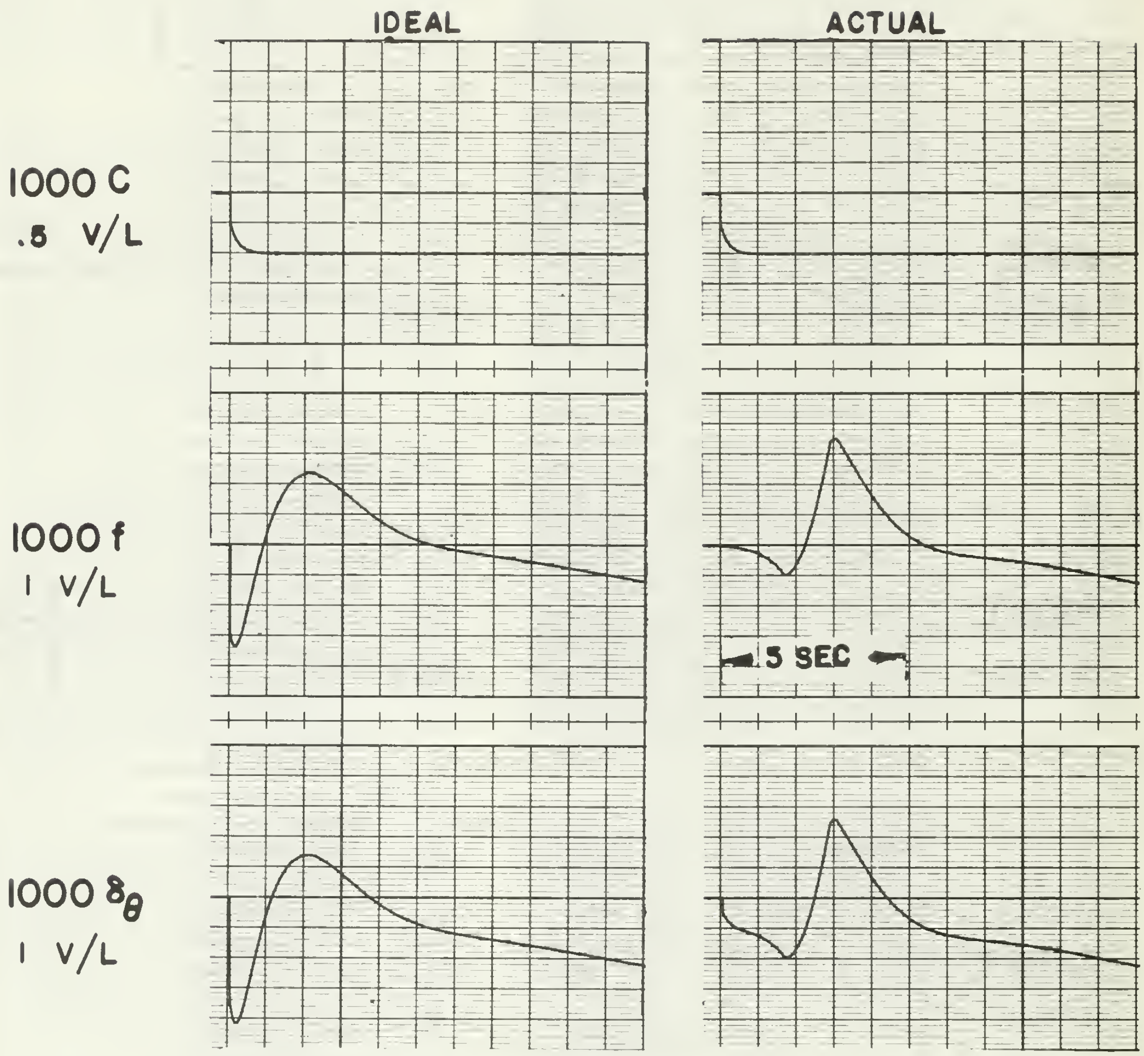

FIGURE 15. CYCLIC INPUTS AT HOVER

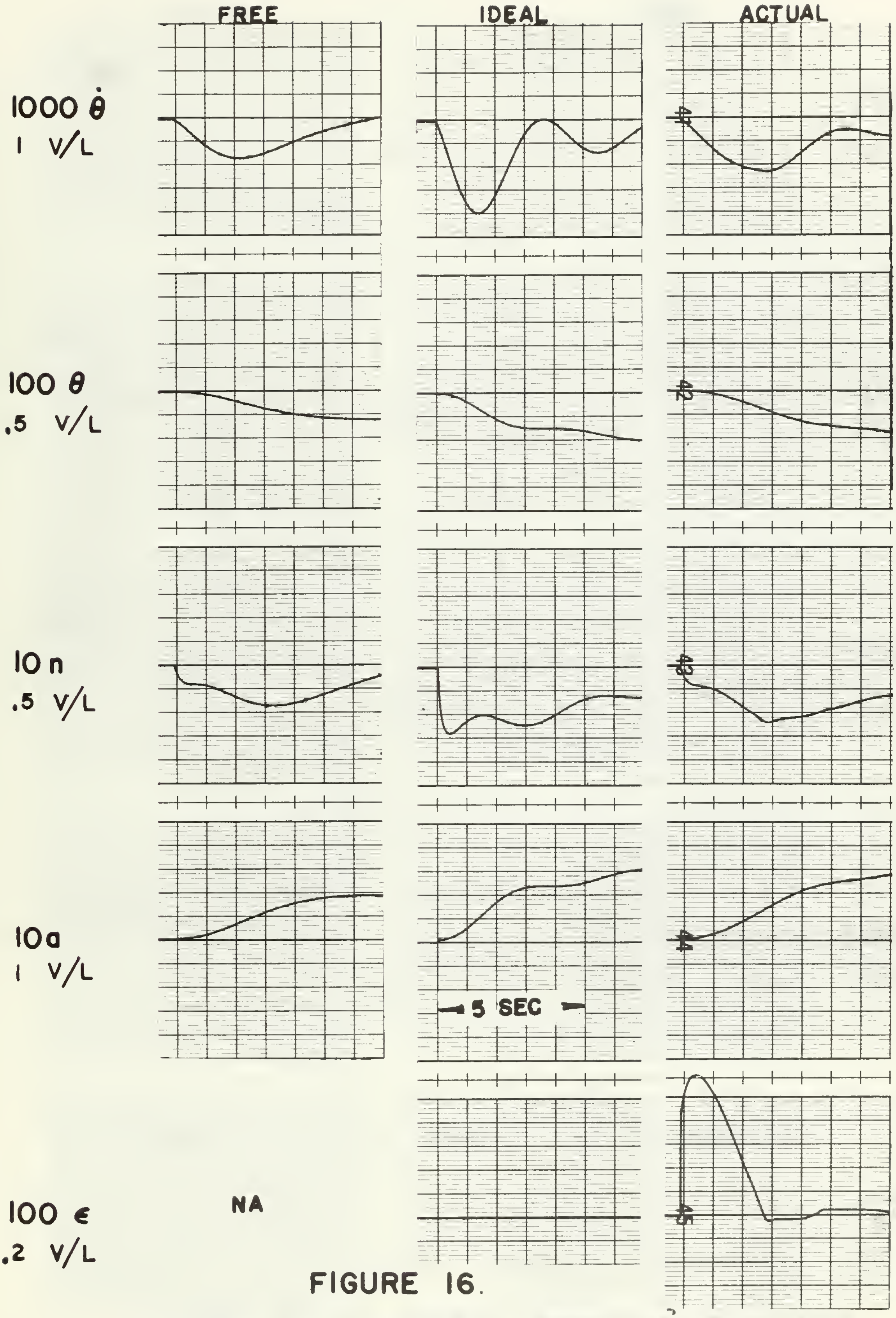

FIGURE 16.

EFFECT OF CONTROLLER AT 40 MPH

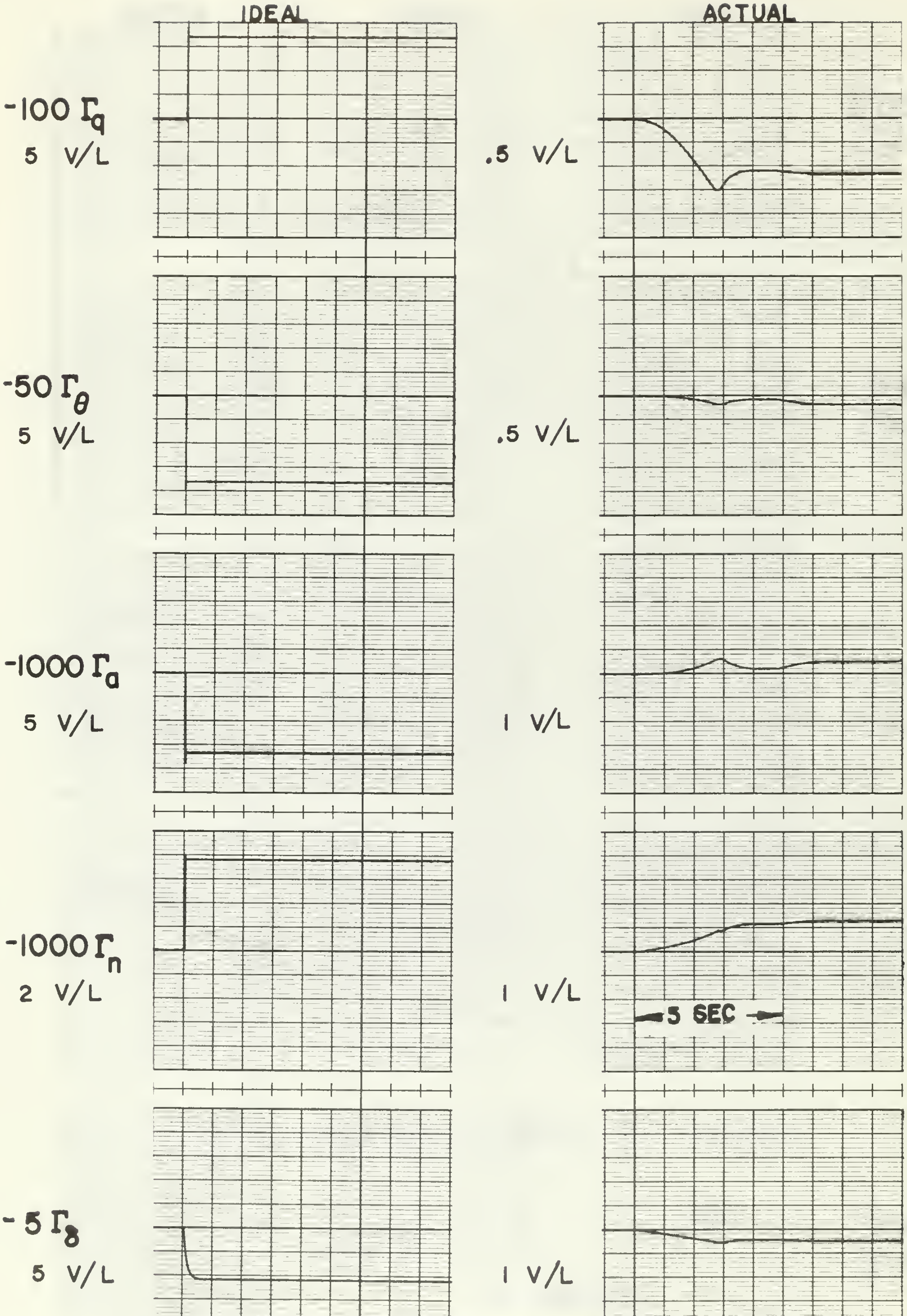

FIGURE 17. VARIABLE GAINS AT 40 MPH

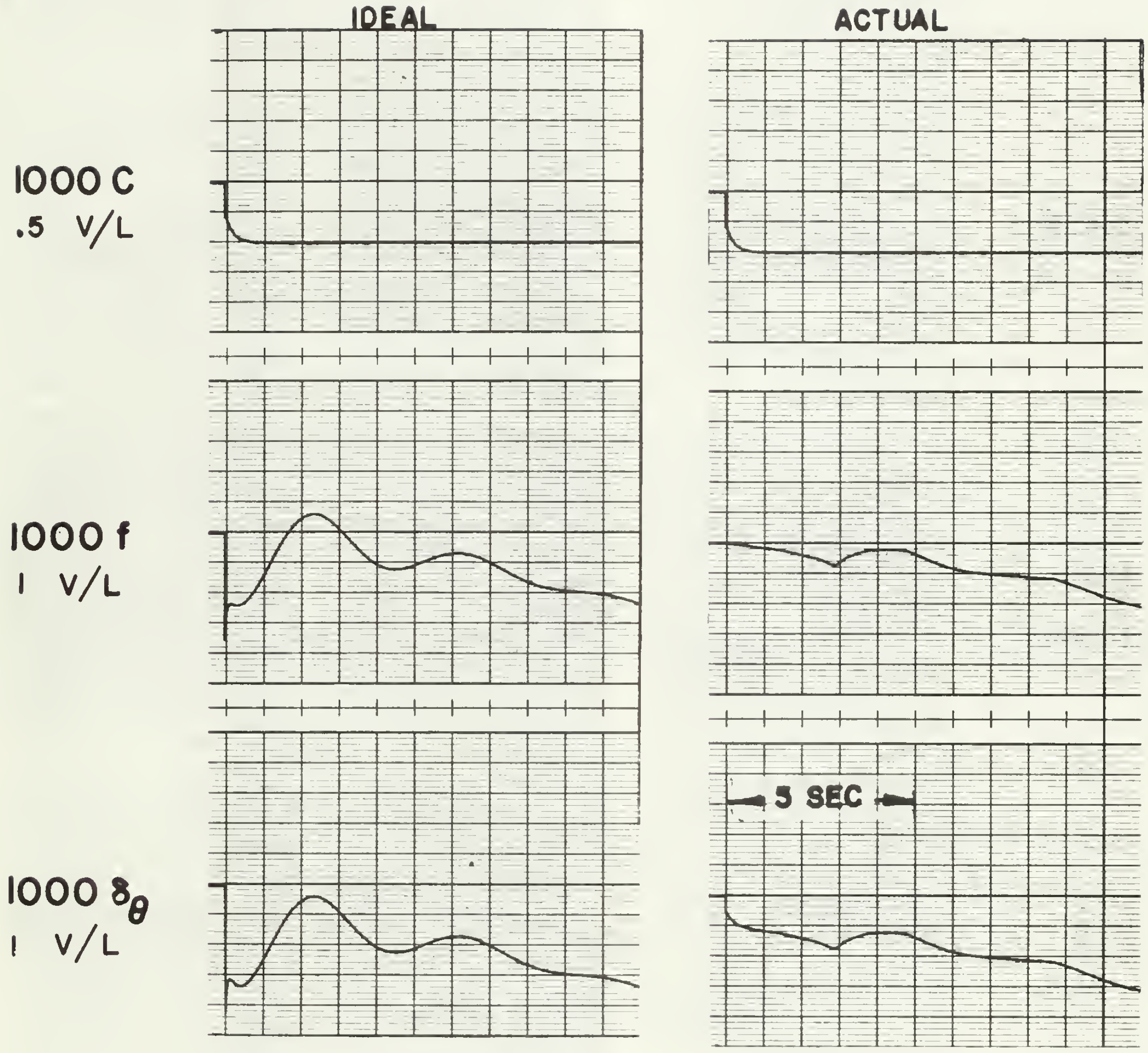

FIGURE 18. CYCLIC INPUTS AT 40 MPH

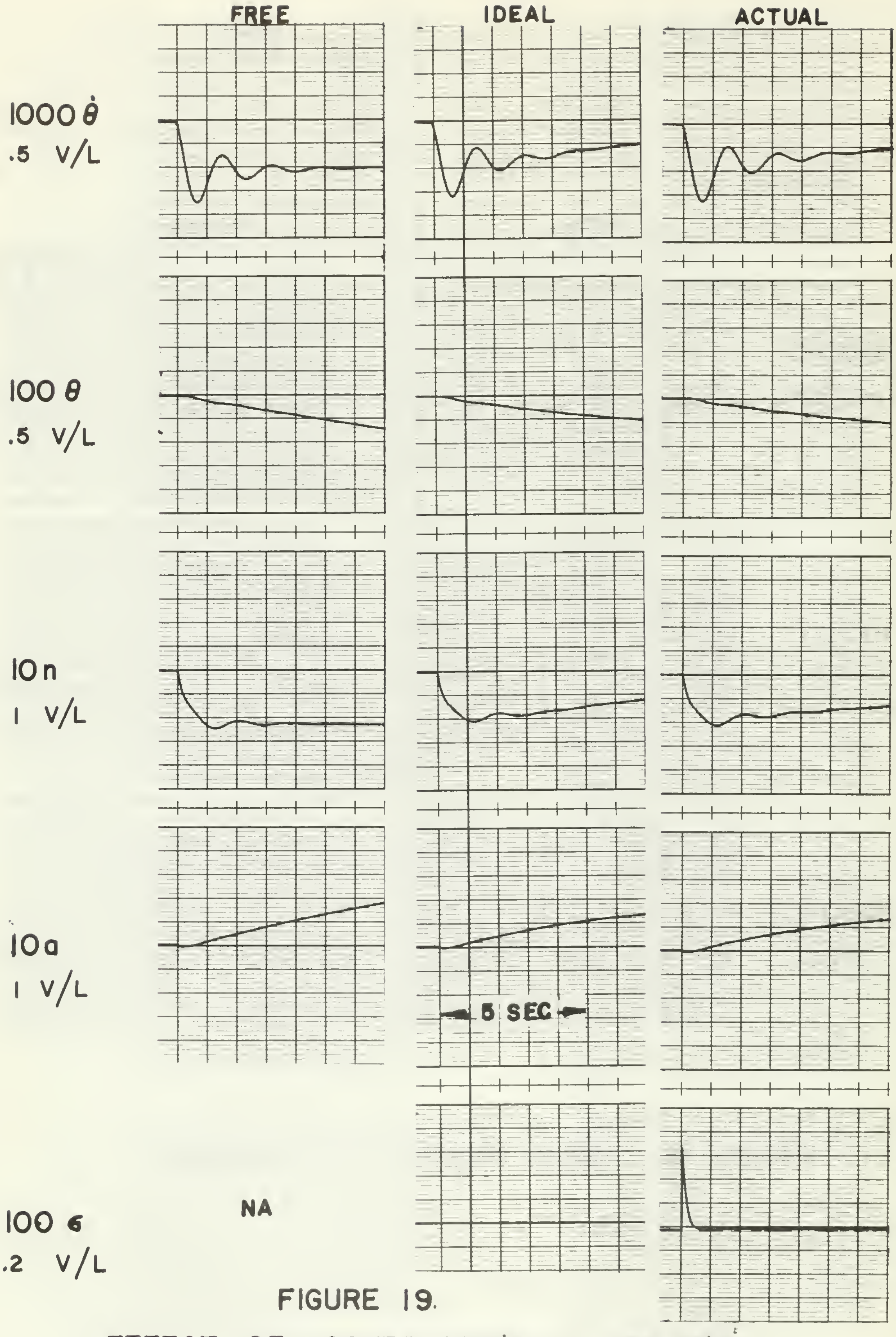

FIGURE 19.

EFFECT OF CONTROLLER AT 140 MPH

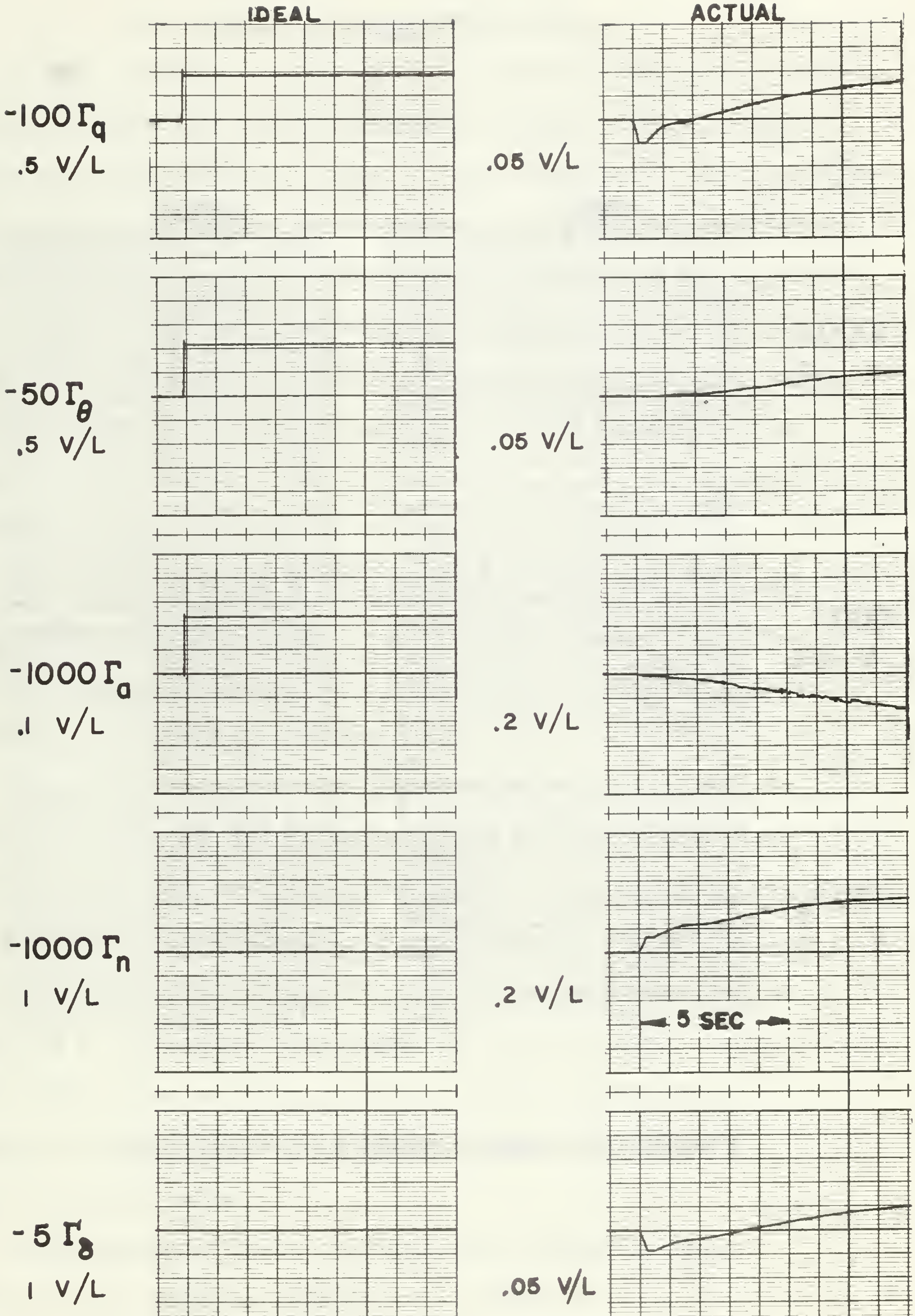

FIGURE 20. VARIABLE GAINS AT 140 MPH

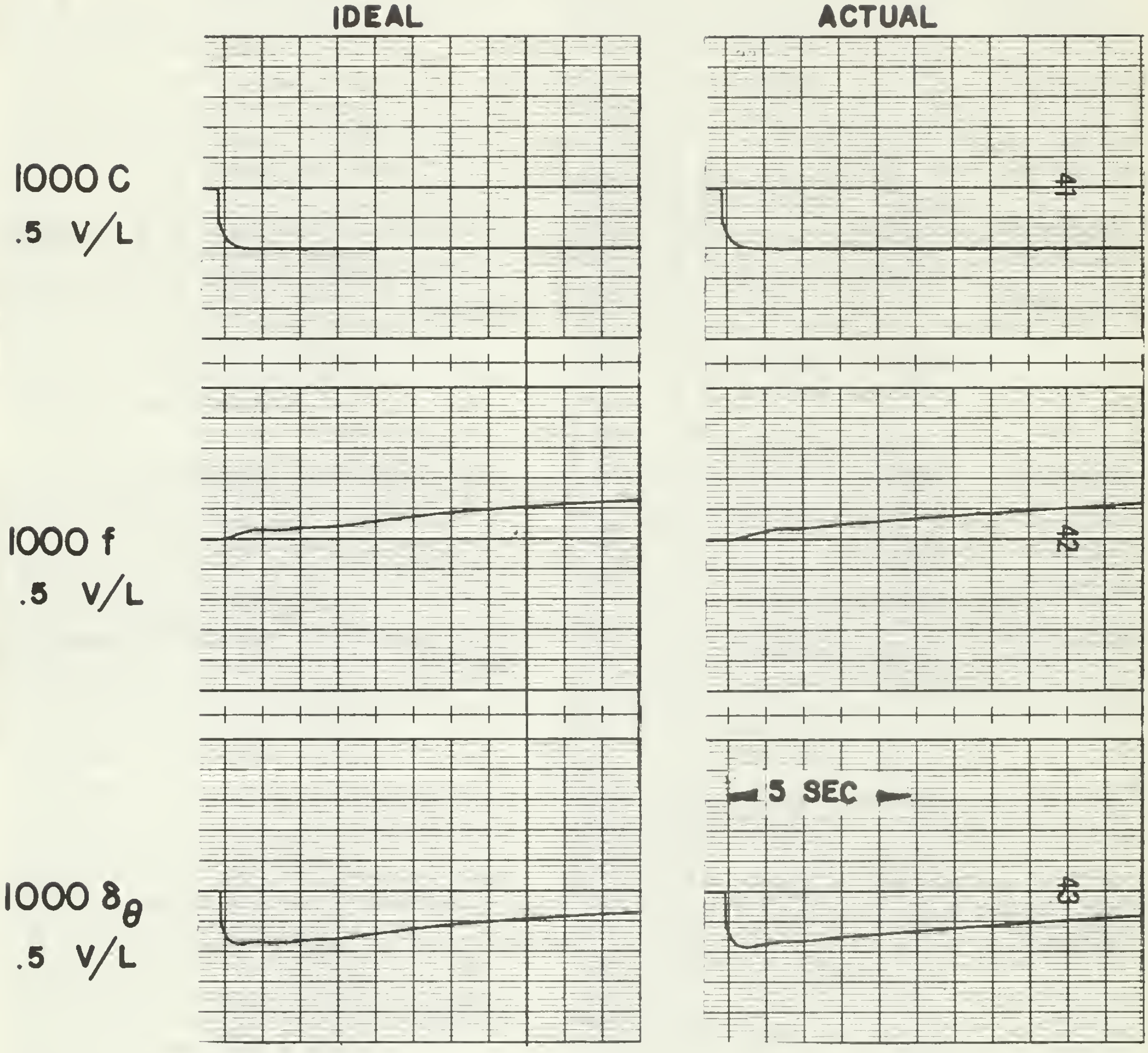

FIGURE 21. CYCLIC INPUTS AT 140 MPH

IV. CONCLUSIONS AND RECOMMENDATIONS

The adaptive controller, based on the equations developed in Chapter II and subject to the listed assumptions and modifications, stabilized the helicopter over the full range of flight conditions. While selection of parameters to stabilize the aircraft did not present any problem, the selection of the proper combination of parameters to both stabilize and to produce desirable handling qualities was extremely difficult. It was not conclusively shown that the handling qualities could be controlled as desired by variation of the β-parameters alone. It is possible that complete control of the handling qualities will require the addition of an outside control loop using fixed gains.

The feedback required to drive the error toward zero was a weak function of the variable gains, which simplified selection of the fixed K-parameters. The controller will behave as desired by using a wide range of fixed K-parameters.

Successful operation of the system was determined primarily by the value of $\bar{\beta}_\delta$ which was a function of the aircraft stability derivatives. Only aircraft in which the sign of $\bar{\beta}_\delta$ is constant over the range of flight conditions would be receptive to this type of system. The magnitude of $\bar{\beta}_\delta$ must also be large enough to keep the magnitude of the variable gains within acceptable limits. An investigation of the magnitude and sign of $\bar{\beta}_\delta$ for several aircraft would be helpful in determining if the shift in sign was an isolated case or normal.

Slight variations in the β-parameters resulted in large changes in the responses at hover and at 40 mph, with the responses at 140 mph being much less sensitive. It was shown

that β_a could be set at zero without great adverse errect in the handling qualities. Further investigation into the control of handling qualities is required. Evaluation of different models, in order to determine whether the variation in stability among them is normally great, would help to determine if the difficulties encountered with the OH-5 model were isolated, or common to most helicopters. Adapting to a nonvarying-C^* control scheme, which would require determining the proper C^*-response envelope would be desirable. The insensitivity of the C^*-response to large deviations of the variable gains from their ideal values reported in Ref. 2 offers further encouragement to the possibility of applying the C^*-criterion to helicopters.

Addition of collective pitch terms and the effects of servos and actuators would be required in order to test the controller under realistic conditions. It is suspected that the normal acceleration changes brought about by the addition of the collective inputs would require some major modifications of the values of the β-parameters selected. It is possible that β_n would have to be eliminated and the system controlled through pitch rate, pitch attitude and the combination of collective and cyclic inputs. Helicopters are normally limited to low values of normal acceleration, so such a limitation would not be unacceptable.

The effect of the blade-flapping terms should be investigated. It could be expected that increases in the values of the gains required would increase the importance of the blade-flapping terms. The effect of large input commands, with the servo and actuator terms included, should be investigated. It could be expected that limit cycles would occur under certain conditions which would require modifications based on the information presented in Ref. 3. Investigation into the

effects of instrument noise and linkage hysteresis would also be required for complete evaluation. Possibilities of increasing the reliability by use of a self-organizing adaptive controller as outlined in Ref. 4 might also be useful for increasing the reliability of the system. Assuming that the problems found in the longitudinal controller could be solved, further development of the system to include the lateral equations of motion would be necessary. Control of lateral motion could be accomplished by the same technique used for the longitudinal system. Control of the directional motion would probably not present large problems if the longitudinal and lateral motions were under control.

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1. ORIGINATING ACTIVITY (Corporate author) Naval Postgraduate School Monterey, California 93940	2a. REPORT SECURITY CLASSIFICATION Unclassified
	2b. GROUP

3. REPORT TITLE
The Extension of an Adaptive Aircraft Control Concept to Helicopters

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)
Master's Thesis; October 1969

5. AUTHOR(S) (First name, middle initial, last name)
John McCoy Hood, Jr.

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
October 1969	59	7

8a. CONTRACT OR GRANT NO.	9a. ORIGINATOR'S REPORT NUMBER(S)
b. PROJECT NO.	
c.	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)
d.	

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13. ABSTRACT

A basic adaptive control scheme for fixed-wing aircraft was modified for use in controlling the longitudinal motion of helicopters. The modification required the addition of two additional feedback variables. Control was applied only to the cyclic pitch input and not to the collective input. It was assumed that a coefficient, the cyclic-pitch control effectiveness, would not change sign throughout the flight envelope.

Analog computer simulation showed that the modified system was capable of stabilizing the model used. The handling qualities of the system were not completely satisfactory and further work is necessary.

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Adaptive Control for Helicopters						
Helicopter Adaptive Controller						
Helicopter Automatic Control						